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PREDESIGN REPORT FOR THE ROTOR SYSTEMS RESEARCH AIRCRAFT

Prepared under Contract No. NAS1-11251 by
BELL HELICOPTER COMPANY
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Fort Worth, Texas

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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INTRODUCTION

This document is submitted in accordance with the requirements of NASA Contract NAS1-11251, Predesign Study for a Rotor Systems Research Aircraft (RSRA). The RSRA is a compound helicopter with a speed ability of 300 KTAS. It incorporates data systems which make possible the in-flight measurement of values normally obtained in wind tunnels.

Bell Helicopter Company (BHC) has conducted a 30-week conceptual predesign of the RSRA. The analysis which led to the definition of the RSRA configuration is described in BHC Report 646-099-001 (A Conceptual Study of the Rotor Systems Research Aircraft). The proposed Development Plan for the RSRA is presented in BHC Report 646-099-003 (A Development Plan for the Rotor Systems Research Aircraft).

This Predesign Report describes the RSRA, including the major subsystems. Detail Specifications and drawings are included.

SUMMARY

This report presents the results of a conceptual predesign of the Bell Helicopter Company's (BHC) candidate for the Rotor Systems Research Aircraft (RSRA). This aircraft was selected by the Government at the end of Part II of the RSRA contract as the better of the two concepts submitted by BHC.

The aircraft is a three-place vehicle in the 24,000-pound gross weight class. It has been designated as the BHC Model 646. It satisfies the RSRA mission requirements as defined in the amended RSRA Statement of Work.

The BHC Model 646 has been predesigned sufficiently to allow an assessment of its performance and stability and control characteristics, and a brief treatment of these subjects is included. The Conceptual Study Report should be referenced for detailed discussions of these subjects.

Major portions of the predesign effort are presented, including: the control system, vibration attenuation system, special data systems, wing assembly, drive train, landing gear, auxiliary engine mount, rotor and auxiliary thrust engines, cockpit arrangements, and a variable geometry rotor. A detailed aircraft specification, based on MIL-STD-832, is presented as Appendix 1. Drawings of the vehicle and major subsystems are also presented as Appendix 2.

A Reliability and Safety Analysis was performed, based on the predesign drawings, and the written definition of the vehicle which was disseminated at BHC for purposes of cost estimation and schedule determination. A Reliability and Safety Analysis was also performed for the draft vehicle specifications. The recommendations of both of these analyses have been incorporated into the RSRA design and specifications.

AIRCRAFT DESCRIPTION

The BHC RSRA is a compound helicopter in the 24,000-pound design gross weight class, with a maximum level-flight speed of 300 KTAS at sea level. Wings and thruster engines are removable to produce a pure helicopter which has a maximum gross weight of 19,210 pounds and an empty weight of 13,252 pounds.

The main rotor is a 55-foot diameter derivation of the BHC Model 240 4-bladed, gimbaled rotor. Blade twist has been decreased from minus eight degrees to zero, and radius has been increased 0.5 foot. The tail rotor is an unmodified BHC Model 240.

An electronic control system which has a full-time mechanical backup is used to control the rotor and fixed-wing control surfaces. This system accepts computer or pilot inputs and controls a dual hydraulic system which powers the rotor and fixed-wing controls. The control system allows "phasing out" of main and tail rotor controls for high-speed operation.

The wing is sized to support the vehicle design gross weight at 150 KTAS at sea level. It will support the design gross weight at 111 KTAS with full-span flaps deployed. Inboard split flaps on the upper trailing edge provide rotor download, and in conjunction with the inboard lower split flaps, provide high drag. The outboard upper split flaps are used only for roll control; they are linked to the cyclic stick. Wing incidence is variable over a range of ±20 degrees.

The main landing gear attaches to trunnions mounted in the fuselage, and retracts forward into the fuselage. The strut-wheel assembly is so constructed that the wheel motion on landing is vertical, even though the struts are canted outboard.

The tail wheel is partially covered by the ventral fin, the lower portion of which pivots with the tail wheel for steering. The landing gear is designed for a 10-feet per second sink rate and a deceleration of 8-feet per second per second.

The main rotor transmission is rated at 3600 SHP. Rotor mast tilt may be ground adjusted to any position between 4 and 12 degrees forward. Rotor power is supplied by two T55-L-7C engines driving through a combining gearbox, from which power for the main and tail rotors, hydraulic pumps, and AC electrical generator is provided.

Two F102-LD-100 turbofan engines provide auxiliary thrust. They are mounted on pylons attached to the fuselage, which are canted upward to lessen wing-engine interference. The engines and pylon are removable to leave a nearly flush fuselage area.

An escape system is provided which consists of rotor severance and canopy removal functions, in conjunction with a YANKEE crew extraction system. Rotor severance and/or canopy removal are available to the crew as independent options.

Table I provides a more detailed description of the RSRA.

A group weight statement is provided in Table II. Mission gross weights, for the primary high-speed design mission and a 30-minute alternate high-speed mission, are delineated in Table III. Center-of-gravity and inertia data are given in Table IV.

TABLE I. AIRCRAFT DESCRIPTION

	Units	Quantity
Design Weights		
Compound		
Empty	lb	17,954
T.O. (Design Mission)	lb	23,738
Maximum	lb	24,761
Pure Helicopter		
Empty	lb	13,252
Maximum	lb	19,210
Fuel Capacity		
Compound	lb	5,063
Pure Helicopter	lb	3,358
Fuselage Dimensions		
Length	ft	51.9
Width	ft	5.25
Height	ft	9.60
Landing Gear		
Main		
Tread	in.	112
Tire Size	in.	26.6 x 6
Oleo Strut Travel	in.	12
Tail Wheel		
Tire Size	in.	18 x 4.4
Oleo Strut Travel	in.	4.7
Axle Travel	in.	12.0
Main Rotor		
Type	--	Modified BHC Model 240 Gimbaled
Number of Blades	--	4
Diameter	ft	55
Chord	in.	25
Solidity	--	.094
Twist	deg	0

TABLE I (Continued)

	Units	Quantity
Main Rotor (Continued)		
Airfoil	--	Mod. Wortmann FX090
Mast Tilt Range	deg	-4 to -12
Collective Range	deg	18
Flapping Range	deg	±10
Wing		
Area	sq ft	225
Aspect Ratio	--	4
Taper Ratio	--	2.0
Sweep Angle (Quarter Chord)	deg	2.85
Incidence Range	deg	±20
Airfoil	--	653A618
MGC	in.	90
MGC Location	in. B.L.	90
Inboard Wing Panels		
Type (Upper and Lower)	--	T.E. Split
Percent of Chord	--	30
Percent of Span	--	60
Maximum Deflection	deg	±60
Area	sq ft	118.0
Outboard Wing Panels		
Percent Chord	--	30
Percent Span	--	40
Maximum Deflection	deg	
As Flaps (Lower)	deg	30
As Ailerons (Upper)	deg	30
Area	sq ft	21.7
Horizontal Stabilizer		
Type	--	Stabilator
Span	ft	14.1
Area	sq ft	50
Aspect Ratio	--	4.0
Taper Ratio	--	2.0
Sweep Angle (Quarter Chord)	deg	2.86
MGC	in.	43.8
MGC Location	in. B.L.	38.0
Airfoil	--	NACA 0012
Incidence Range	deg	±15

TABLE I (Continued)

	Units	Quantity
Vertical Stabilizer		
Area	sq ft	21.2
Aspect Ratio	--	1.7
Taper Ratio	--	2.5
Sweep Angle (Quarter Chord)	deg	47
MGC	in.	47
MGC Location	in. B.L.	0
Airfoil	--	NACA 0012
Rudder		
Area	sq ft	4.3
Percent Chord	--	30
Percent Span	--	50
Incidence Range	deg	±30
Vertical Fin		
Area	sq ft	17.5
Aspect Ratio	--	0.75
Tail Rotor		
Type	--	BHC Model 240
Number of Blades	--	4
Diameter	ft	10.0
Solidity	--	0.191
Twist	deg	0
Airfoil	--	FX083
Main Rotor Engines		
Type	--	T55-L7-C
Mil. Power (Uninstalled)	shp	2 x 2650
Mil. Power (Installed)	shp	2 x 2520
Auxiliary Engines (Installed)	--	F102LD100
Mil. S.L. Static Thrust	lb	2 x 7200
Mil. S.L. Thrust at 300 KTAS	lb	2 x 4100

TABLE II. GROUP WEIGHT STATEMENT

Group	Weight (lb)
Rotor	1921.0
Wing	1209.0
Tail	236.8
Body	2700.0
Gear	805.0
Flight Controls	747.0
Engine Section	1041.8
Propulsion	5806.7
Instruments	119.7
Hydraulics	165.0
Electrical	382.0
Electronics	224.4
Furnishings and Equipment	159.2
Air Conditioning	162.0
Special Provisions	
Pylon Support	280.0
Crew Escape Systems	324.0
Wing Tilt and Balance	350.0
Wing High Lift Systems	150.0
Instrument Packages	80.0
Ballast Provisions	10.0
Rotor Tilt Provision	80.0
Instrumentation	1000.0
Weight Empty	17953.6

TABLE III. HIGH-SPEED MISSION GROSS WEIGHTS
SEA LEVEL

	Primary Mission	Alternate Mission
Weight Empty	17953.6 (lb)	17953.6 (lb)
Useful Load		
Crew (3 Members)	600.0	600.0
Fuel	3040.0	5063.0
Payload	2000.0	1000.0
Engine Oil	53.8	53.8
Engine Oil - Trapped	17.2	17.2
Oil - Transmission and Gearboxes	54.3	54.3
Fuel - Trapped	19.3	19.3
Auxiliary Fuel Instl.	-	-
Mission Gross Weights	23738.2	24761.2

TABLE IV. CENTER OF GRAVITY AND INERTIA DATA

	Weight Empty	Primary Mission	Alternate Mission
Weight (lb)	17954	23738	24761
Center of Gravity (in.)			
Longitudinal	FS 229	FS 222	FS 227
Lateral	BL 299	BL 299	BL 299
Vertical	WL 98	WL 92	WL 91
Inertia (slug-ft^2)			
Roll	13719	15066	17226
Pitch	50414	55707	54186
Yaw	51141	55664	55790
I_{XZ}	524	1539	588
Principal Axis Inclination	0.8°	2.17°	0.87°

SPECIAL DATA SYSTEMS

GENERAL

The data requirement for the RSRA includes all data systems normally associated with wind-tunnel testing of rotary and fixed-wing aircraft, plus force and moment measurements peculiar to the RSRA. Data systems are presented in three groupings: main rotor (excluding blades), main rotor blades, and "other" measurements.

REQUIREMENTS

The following measurements are necessary to provide the data required by the RSRA mission:

Main Rotor (Excluding Blades)

- Rotor RPM and Azimuth
- Hub Moment
- Torque
- Lift
- Drag
- Side Force
- Lateral and Longitudinal Cyclic Stick Position
- Collective Stick Position
- Boost Tube Positions
- SCAS Input

Main Rotor Blades

- Feathering
- Flapping
- Chordwise Pressure Distribution
- Flow Direction
- Angle of Attack
- Chordwise Bending Moment
- Flapwise Bending Moment
- Chordwise Acceleration
- Flapwise Acceleration

Other Measurements

- Wing Incidence
- Wing Chordwise and Beamwise Forces
- Tail Rotor Shaft Force
- Stabilator Beamwise Force
- Auxiliary Engine Thrust

- Aircraft State

    Altitude
    Airspeed
    Fuselage Angle of Attack
    Fuselage Sideslip Angle
    Three-Axes Angular Accelerations and Rates
    Three-Axes Linear Accelerations
    Roll and Pitch Attitude

Redundancy Requirements

Several of the measurements listed above are used by the automatic flight control system to control the aircraft. Rotor flapping, airspeed, fuselage angle of attack, fuselage motion rates, and fuselage attitudes are considered to be critical and will require redundant measurements.

MAIN ROTOR MEASUREMENTS (EXCLUDING BLADES)

Rotor RPM and Azimuth

The simplest means of measuring these values is by means of a transducer located between the mast and transmission case, but this results in an error caused by mast twist under torque. To avoid mast twist error, the transducer is placed atop the nonrotating standpipe which is located inside the hollow mast. Toothed wheels and magnetic pickups are avoided, since their signal accuracy is dependent on RPM. The signal from these devices becomes unstable at small rotational speeds. The best solution is to use a photocell/slotted wheel-type shaft encoder. Two outputs are available from such devices: a one/rev pulse aligned with the master blade for synchronizing data, and a 512 pulse/rev signal for exact measurement of rotor azimuth. The number 512 represents $(2)^9$, which provides a sampling rate compatible with Fourier Transform Methods of Analysis.

Hub Moment, Mast Torque, Lift, Drag, and Side Force

The measurement of all these values is the function of the main rotor balance system. BHC is considering two candidates for this function: the focused pylon and the mast balance.

The focused pylon may be used as a main rotor balance by measuring the axial force in the focusing legs, in the torque restraint links, and in the longitudinal and lateral restraint links. This device may be seen in Drawing 646-010-400. A sketch of a focused pylon is included as Figure 1.

The internal mast balance system is a device which is positioned inside the mast by means of bearings which fit against the inside walls of the mast. The device measures bending moment in orthogonal directions at two vertical locations. Lift and torque cannot be

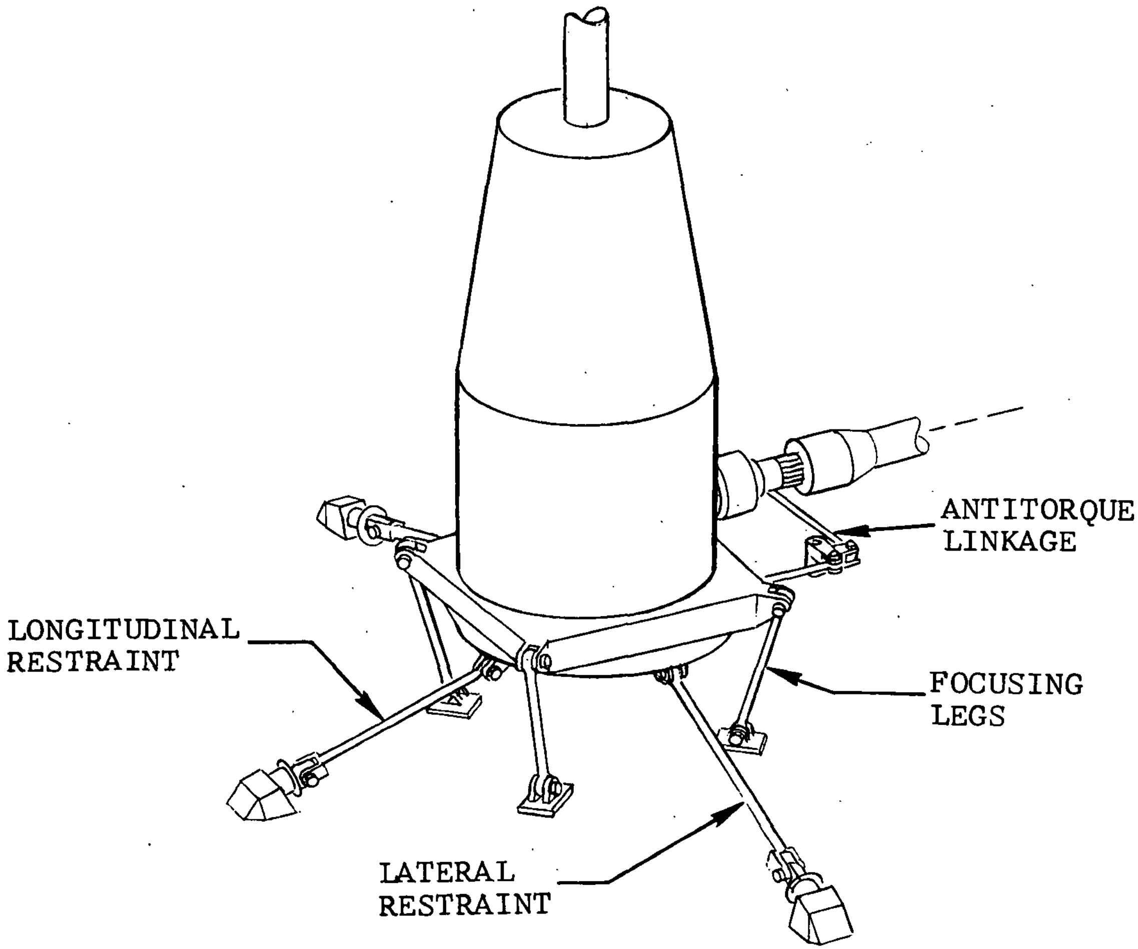


Figure 1. The Focused Pylon.

measured by such a device, but lift measurement may be obtained by instrumenting either the Servo-Null arms or the focused pylon legs, and torque may be obtained from the torque restraint links. Drawing 646-010-500 describes the mast balance system. A sketch of one element of this system is included as Figure 2.

Regardless of which system is used, strain gages are used as transducers.

Position Measurements (Cyclic Stick Position, Collective Stick Position, Boost Tube Positions, and SCAS Inputs)

Measurements of the first two values are critical, since they are used in the determination of other control functions by the automatic flight control system. Potentiometers may be used, but do not meet accuracy requirements unless great care is taken with the design of their mounts. Selsyn units provide longer life and good accuracy, but are costly -- particularly if conversion to a digital output is required. Discrete shaft encoders could provide direct digital output, but are too large to be practical. The best selection is to use Selsyn units for measurements which require accuracies as great as ±0.5 percent. Analog to digital conversion may be provided for these units to allow direct computer input. This system provides good accuracy and small size. Linear potentiometers are selected for other functions.

Table V summarizes the types of measurement devices used and their expected accuracies. The accuracies shown for the mast balance are based on a recently conducted BHC analytical study. A NASA-funded project is presently underway to investigate the focused pylon as a main rotor balance. The accuracies shown for the focused pylon balance are estimates from previous BHC studies.

TABLE V. DATA SYSTEM-MAIN ROTOR EXCLUDING BLADES

Item Measured	Type Transducer	Est. Accuracy (% Full Scale)
Rotor RPM and Azimuth	Shaft Encoder	±1.0 Deg
Hub Moment	Strain Gage	±7.0%**
Mast Torque	Strain Gage	5-10%*
Lift	Strain Gage	5-10%*
Drag	Strain Gage	±2.0%**
Side Force	Strain Gage	±2.0%**
Cyclic Stick Position	Selsyn	±0.5%
Collective Stick Position	Selsyn	±0.5%
Boost Tube Positions	Linear Pot.	±1.0%
SCAS Input	Linear Pot.	±1.0%

*Focused pylon in conjunction with vertical vibration attenuation system.

**Mast balance in conjunction with vertical vibration attenuation system.

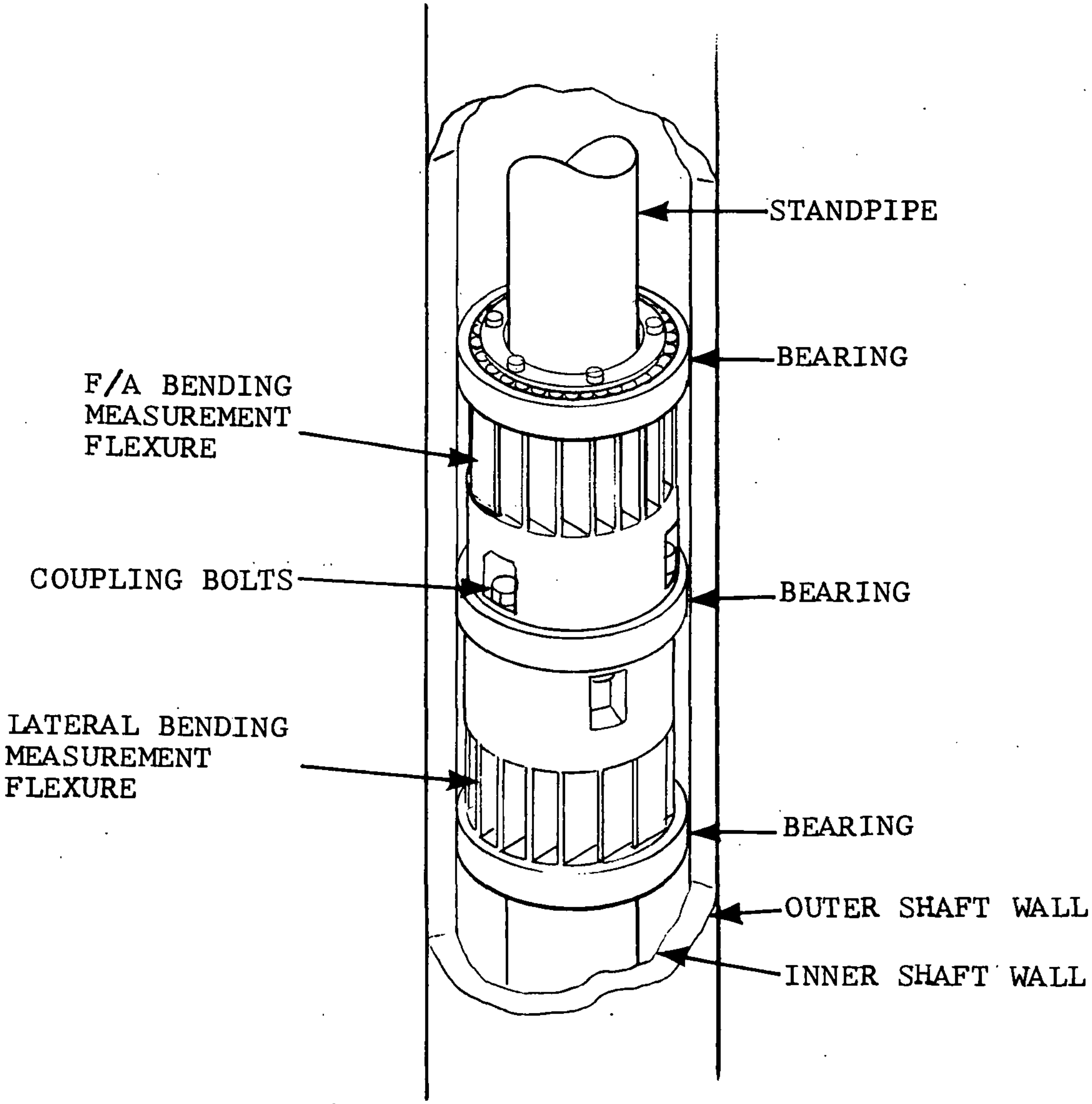


Figure 2. Mast Balance System.

MAIN ROTOR BLADE MEASUREMENTS

Feathering and Flapping

Since these measurements are made in the rotating system, Selsyn units are required to provide good accuracy and long life.

Main Rotor Blade Environment (Pressure Distribution, Flow Directions, Local Angles of Attack, and Chordwise and Flapwise Bending Moments and Accelerations)

BHC has designed and tested (under U. S. Army Contract DAAJ02-70-0036) an advanced instrumentation system for rotor blades which allows measurement of these values. The system requires further development, particularly in the measurement of pressure distribution and the multiplexing of signals from the rotating system, but it is considered to be the best available means of measuring these values.

The BHC Advanced Instrumentation System provides for the measurement of the following values by means of transducers placed in an 0.1-inch thick honeycomb and fiber glass glove which is formed over the rotor blades:

- Pressure Distribution:

  Pressure gages are used to measure absolute pressures on the upper and lower surface at eleven chord stations and at five radial stations. All remaining instrumentation is installed on the opposite blade.

- Flow Direction:

  A Boundary Layer Button (BLB) is a BHC patented device which uses two orthogonally mounted differential pressure transducers to measure flow direction. These devices will be installed on the upper and lower blade surfaces at 30-, 60-, and 90-percent chord positions at five radial stations.

- Local Angle of Attack:

  The leading-edge hot wire anemometer has proven to be a suitable means of measuring angle of attack. BHC has developed equipment, located in the rotating system, which determines the wire of maximum voltage (stagnation point), eliminating the requirement to carry all hot wire signals through the slip rings. Hot wire anemometers are placed at each of the five radial stations.

- Bending Moments and Accelerations:

  Two strain gages and one biaxial accelerometer is installed at each of eleven radial stations.

- Multiplexing and Data Acquisition:

  Transmitting the quantity of data measured by the above transducers from the rotating to nonrotating system using a slip ring is not practical. Therefore, the signals are multiplexed in the rotating system. The BHC multiplex system requires:

  $N = 9 + \frac{T}{13}$ slip ring elements, where N is the total number of slip ring elements required, and T is the number of transducers.

  This multiplexing system has been tested, but more development is required before satisfaction is assured.

Table VI summarizes the types of measurement devices used and their expected accuracies.

TABLE VI. DATA SYSTEM-MAIN ROTOR BLADES

Item Measured	Type Transducer	Est. Accuracy (%Full Scale)
Feathering	Selsyn	±0.5%
Flapping	Selsyn	±0.5%
Chordwise Pressure Distr.	Diff. Pressure Gages	±1.0%
Flow Direction	Boundary Layer Button	±1.0%
Local Angle of Attack	L.E. Hot Wire Anemometer	±1.0%
Chordwise Bending Moment	Strain Gage	±1.0%
Flapwise Bending Moment	Strain Gage	±1.0%
Chordwise Acceleration	Biaxial Accelerometer	±1.0%
Flapwise Acceleration	Biaxial Accelerometer	±1.0%

OTHER MEASUREMENTS

Wing incidence is measured by a Selsyn transducer mounted directly to the wing incidence control arm.

Wing chordwise and beamwise forces and tail rotor shaft axial force are measured by flexural balances specially designed for these uses.

The wing balance weighs about 325 pounds and has spring rates of 15.6×10^6 lb/in. (beamwise) and 6.43×10^6 lb/in. (chordwise). Although each flexural balance must be calibrated before accuracy can be precisely stated, BHC experience with this type of balance indicates that accuracies of 2 to 3 percent are normal. Drawing 646-020-100 depicts the wing balance.

BHC has tried a variety of methods for measuring tail rotor shaft axial force by applying strain gages to existing aircraft structure, but none was satisfactory. The flexural balance was chosen as the only logical system, even though it is capable of providing much more information than just the measurement of a single component. Drawing 646-010-800 depicts the tail rotor balance.

The placement of strain gages on the stabilator spar measures only spar moment, not beamwise force, but a reasonable correlation is routinely achieved in BHC test work. The measurement of this value serves the primary purpose (in rotor tests) of providing a backup for the main rotor balance, and the accuracy attainable with a strain gage installation is considered adequate.

Auxiliary engine thrust is measured by a straightforward strain gage installation on the main engine support and drag brace (see Drawing 646-060-200).

Aircraft State (fuselage angle of attack, sideslip angle, altitude, airspeed) - The measurement of airspeed below a speed of approximately 20 knots is historically a problem. The presence of gusts and vibration of the instrumented boom also introduces measurement errors in determination of fuselage angle of attack and sideslip. The least complex method of measuring airspeed and flow angles is by means of a pitot tube and vanes, but the best accuracies which may be obtained with these devices is 5 and 3 percent respectively, even at moderate values of dynamic pressure.

Better accuracies may be obtained with differential pressure sensing schemes, such as the "Rosemount" system. It is reasonable to expect angular measurements to within ±0.25 degrees, and airspeed measurement to within ±2.0 knots (above 20 knots), if this system is used. Altitude is also measured by this system to within ±50 feet.

A Rosemount Model 858AJ-28 flow angle sensor with Rosemount 831BA and 830BA pressure sensors are selected.

Accelerations, Angular Velocities, and Attitudes

The classic approach to these measurements is with accelerometers and gyros providing separate outputs. Accuracies of 3 percent can be expected and most units provide a high level signal output.

Table VII summarizes the selected devices and the accuracies to be expected.

TABLE VII. DATA SYSTEM-OTHER MEASUREMENTS

Item Measured	Type Transducer	Est. Accuracy (% Full Scale)
Wing Incidence	Selsyn	±0.5%
Wing Chrodwise Force	Flexural Balance	±3.0%
Wing Beamwise Force	Flexural Balance	±3.0%
Tail Rotor Shaftwise Force	Flexural Balance	±2.0%
Stabilator Beamwise Force	Strain Gage	±4.0%
Auxiliary Engine Thrust	Strain Gage	±2.0%
Altitude	Rosemount	±50 Ft
Airspeed	Rosemount	±2.0 Kt
Fuselage Angle of Attack	Rosemount	±0.25 Deg
Fuselage Sideslip Angle	Rosemount	±0.25 Deg
3-Axes Linear Acceleration	Force Servo Balances	±1.0%
3-Axes Angular Acceleration	Force Servo Balances	±1.0%
3-Axes Angular Velocities	Rate Gyro	±1.0%
Pitch and Roll Attitude	Attitude Gyro	±0.5%

PERFORMANCE DATA

GENERAL

Predicted performance is presented here to show the performance of the RSRA in the three missions prescribed by the Statement of Work (SOW): hover, high-speed, and helicopter simulation. All three missions are flown in the compound configuration with the specific design four-bladed, gimbaled, 55-foot diameter rotor (solidity = 0.094).

METHODS OF ANALYSIS

Rotor performance was predicted from data presented in NASA CR114 and the Bell Rotor Aerodynamic Method (BRAM) computer program. The BRAM program was checked to assure agreement with CR114 and was used for those areas of rotor performance which CR114 did not include.

Hovering download was computed using download factors as directed by the SOW.

Engine fuel flows were based on the engine manufacturer's data, with fuel flow rates increased five percent and engine inlet losses considered to be two percent for auxiliary thrust engines and five percent for rotor engines.

Parasite drag was estimated by three methods. In addition to the method directed by the SOW, a method commonly used for fixed-wing aircraft design, and the method normally employed by BHC, were also used. The SOW method always yielded higher drag values than the other two methods and was used for all performance estimations shown here.

Main and tail rotor hub drags were estimated by the SOW prescribed method:

$$f_{hub} = 0.07\ (SHP)^{0.58}$$

HIGH-SPEED MISSION

The design high-speed mission includes the following elements:

- 2 minutes warmup and takeoff, all engines at NRP
- 2 minutes flight at minimum airspeed
- climb to cruise altitude
- accelerate to cruise speed
- cruise at 300 KTAS for 15 minutes

- decelerate to descent schedule
- descend to sea level
- hover for 2 minutes OGE
- land with fuel reserve of 10 percent of takeoff fuel

The design payload of 3000 pounds includes 1000 pounds of non-removable instrumentation.

An alternate mission is defined by the requirement to cruise at 300 KTAS for 30 monutes. Payload is reduced to 2000 pounds for this mission, and design maneuvering ability (+4.0 g, - 1.5 g) and landing sink rates (10 fps) do not apply.

The Conceptual Study Report may be referenced for further details concerning performance estimation for the high-speed mission.

Figures 3 and 4 present RSRA high-speed performance at Sea Level and 9500 feet, respectively.

HOVER PERFORMANCE

Hover performance was calculated for the following mission:

- 2 minutes warmup and takeoff, all engines at NRP
- 2 minutes flight at minimum airspeed
- hover for 30 minutes OGE
- fly 10 nautical miles at optimum airspeed
- hover for 2 minutes OGE
- land with 10-percent fuel reserve

Figure 5 depicts hover performance in the compound configuration at Sea Level Standard Day, and at Sea Level, 95°F, with the specific design rotor.

HELICOPTER SIMULATION

The purpose of this mission is to map the performance of test rotors between 100 and 200 knots. Wing lift and download, and auxiliary thrust and drag, are used to maintain a given airspeed. Refer to the Conceptual Study Report for a detailed discussion of the assumptions and calculation methods for this mission.

The best way to state the ability of the RSRA to map rotor performance is to state the maximum size rotor (diameter and solidity) which may be driven to the Upper Stall Limit (USL) at a given control plane angle of attack and airspeed. Since the maximum diameter is established at 55.0 feet, solidity remains as the sole measurement of rotor size.

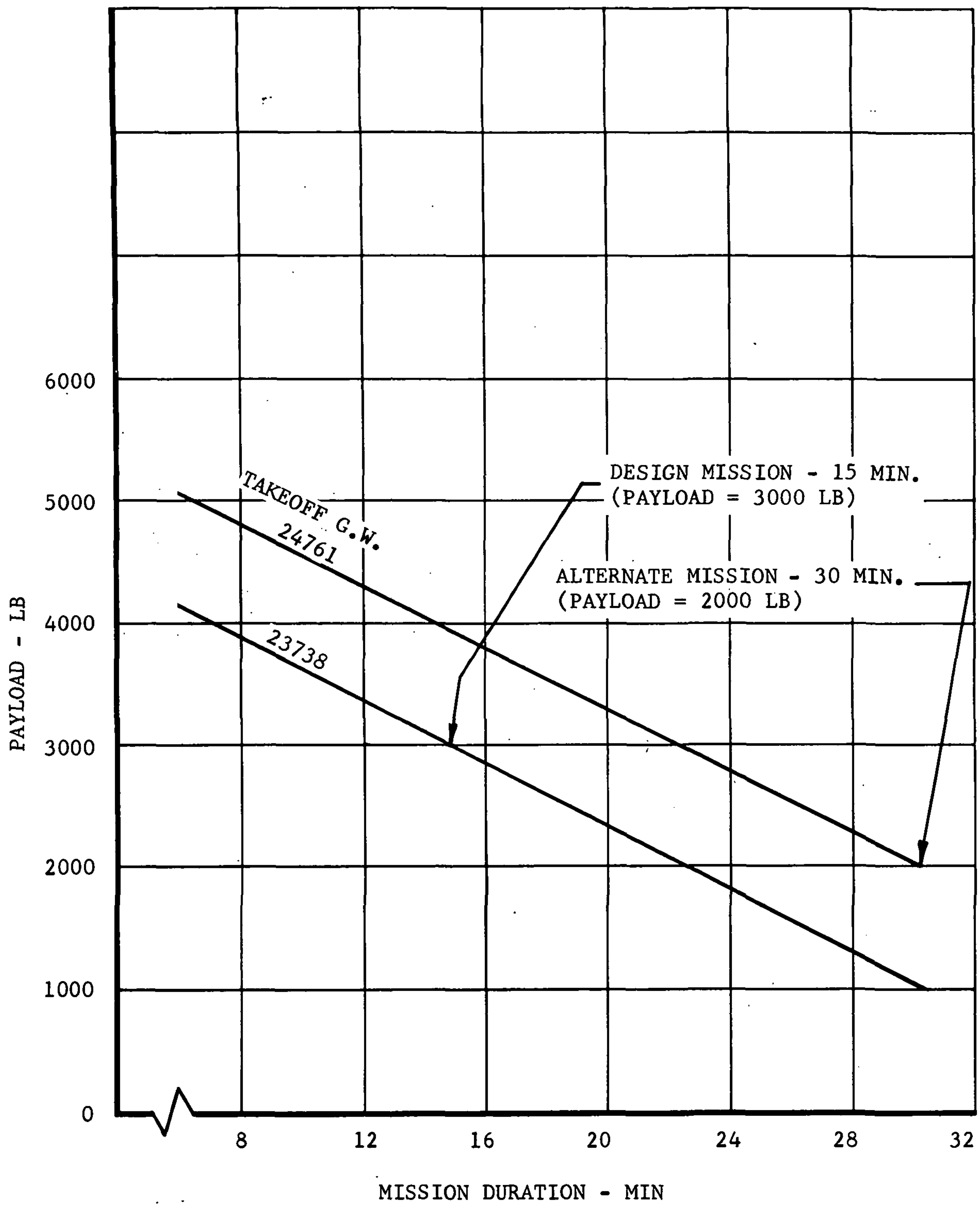


Figure 3. 300-Knot Mission Performance, Sea Level Standard Day.

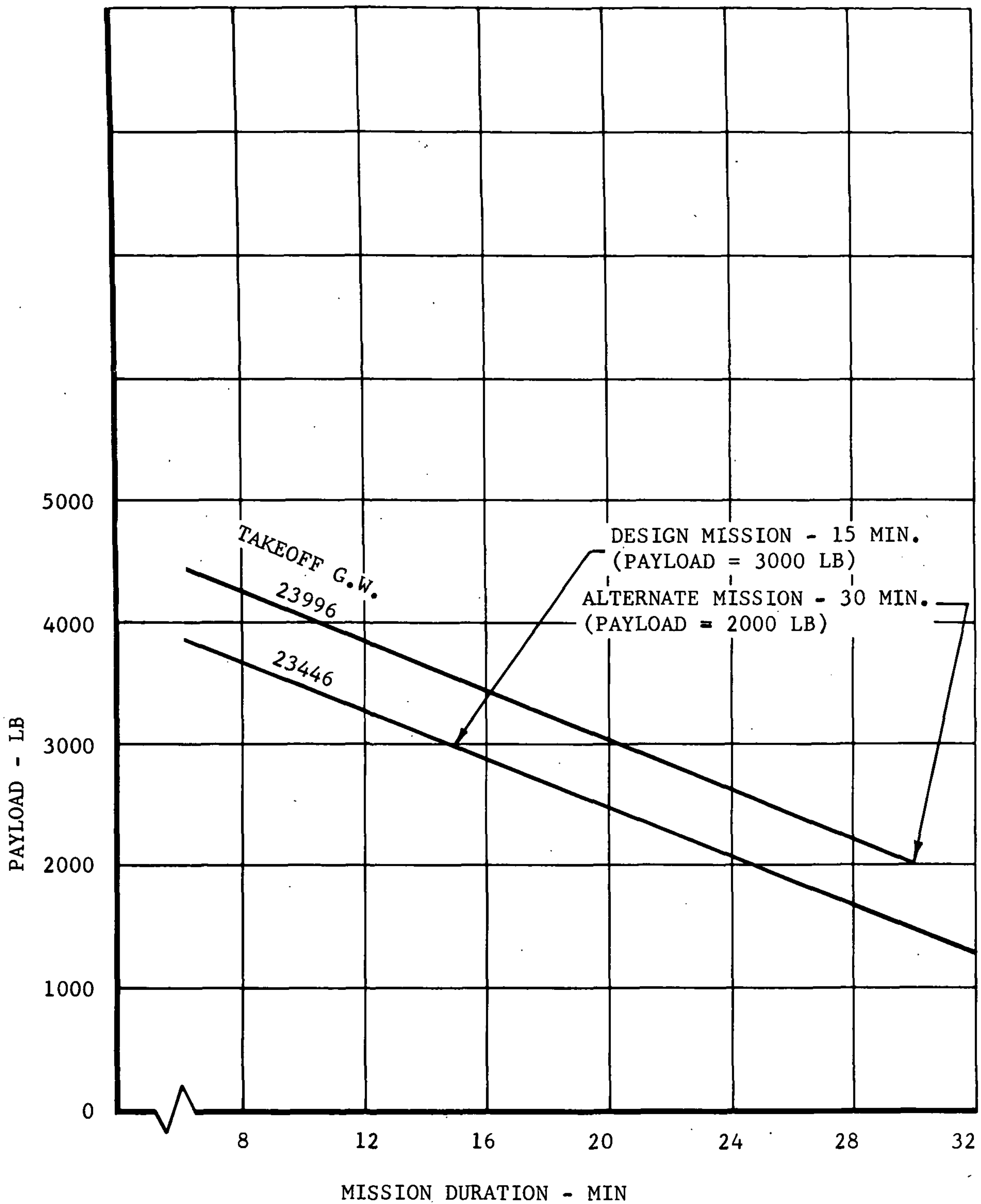


Figure 4. 300-Knot Mission Performance, 9500 Feet Standard Day.

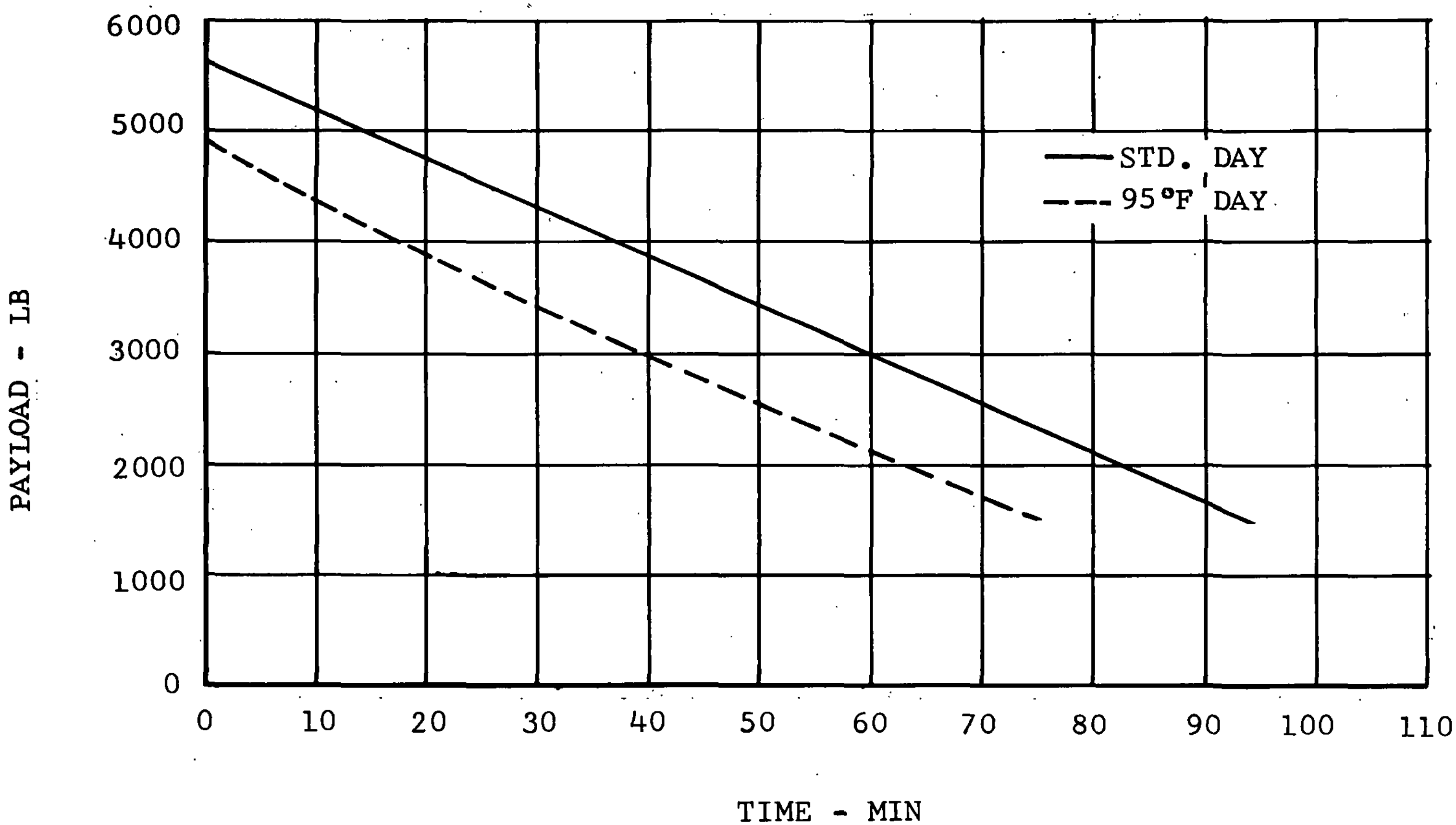


Figure 5. Hover Performance, Sea Level.

Figure 6 depicts the maximum solidity which can be driven to the USL (at 100 KTAS, sea level) as a function of control plane angle of attack. Figure 7 provides the same information for 200 knots.

It may be seen that a large range of rotor solidities can be tested, even at the worst condition of 100 knots. For example, the specific design rotor (solidity = 0.094) may be driven to the USL at any control plane angle of attack between -3 and +9 degrees. At 200 knots the RSRA can test rotor solidities well above practical values. The specific design rotor test limit is determined only by autorotation (+3 degrees).

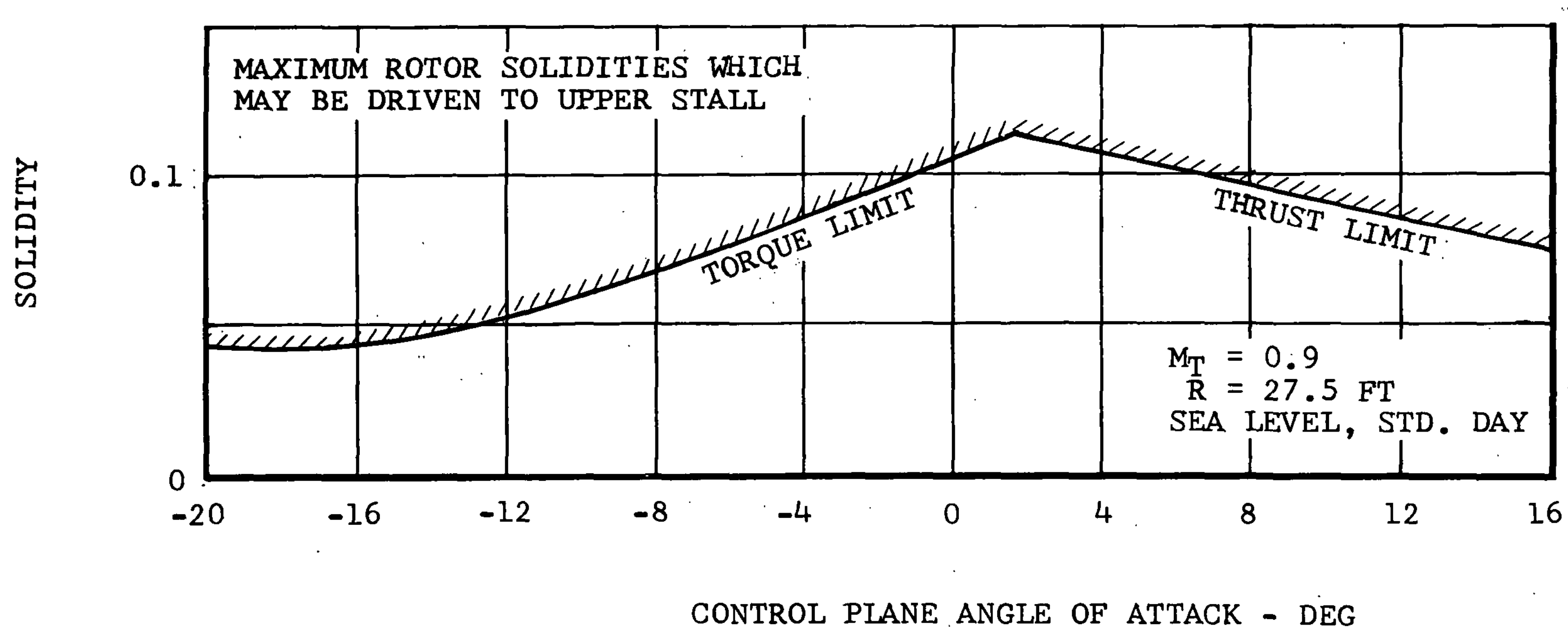


Figure 6. Rotor Test Versatility, 100 Knots.

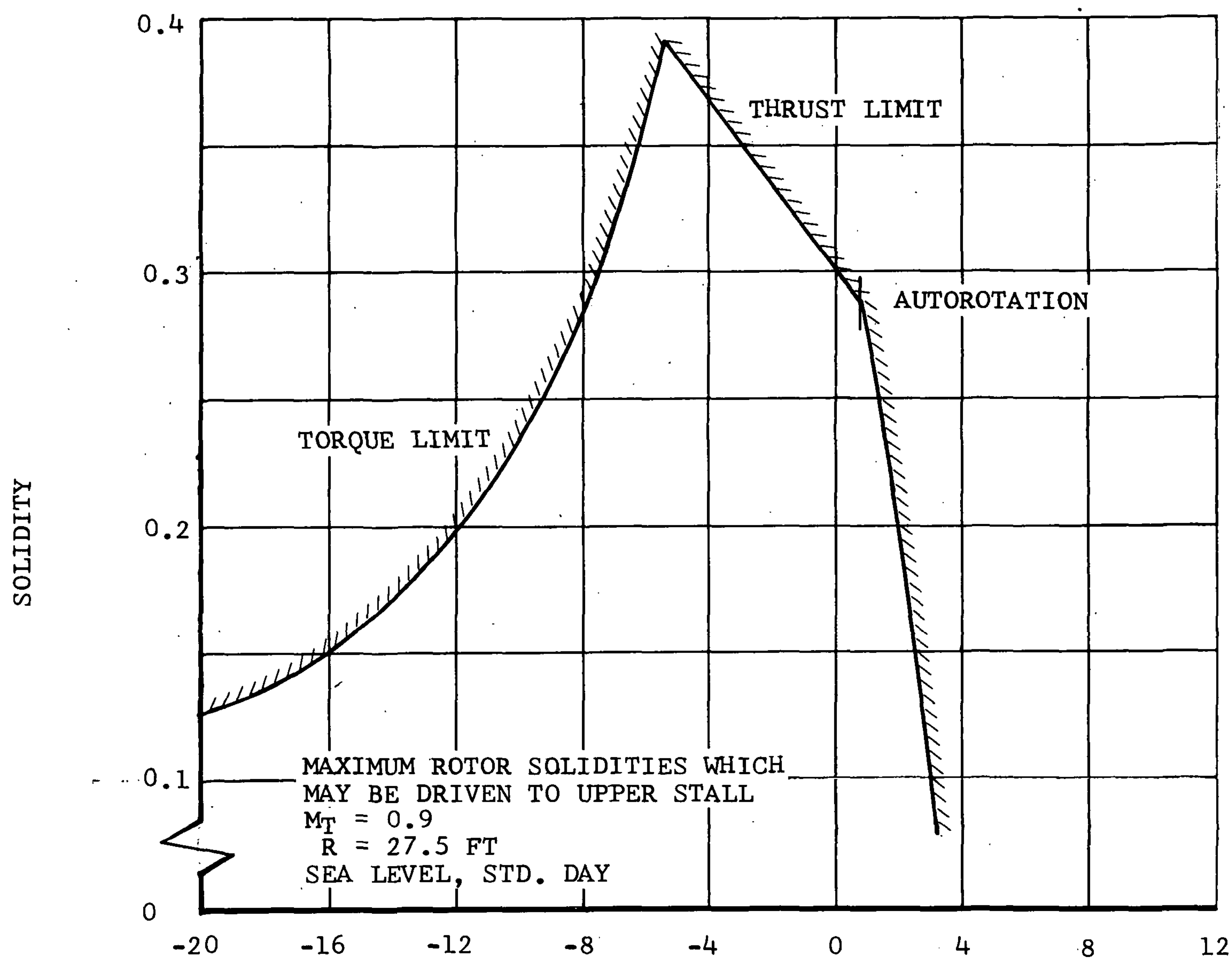


Figure 7. Rotor Test Versatility, 200 Knots.

ROTOR AND AUXILIARY THRUST ENGINES

AUXILIARY THRUST REQUIREMENTS

The RSRA has a flat plate drag area at 300 knots sea level standard day, of 26.8 square feet, requiring a thrust of 8150 pounds to attain that speed. The required thrust at 9500 feet sea level standard day is 6100 pounds. The requirement to allow for a 30-minute cruise at 300 knots sea level standard day, dictated the selection of fan engines. Installed specific fuel consumption as low as .73 lb fuel/lb thrust/hour (300 KTAS sea level standard day) may be obtained for fan engines, versus corresponding sfc's of about 1.2 for turbojet engines. For the RSRA, selection of a fan engine means a fuel weight savings of about 1000 pounds for the 15-minute high-speed design mission, which allows a reduction in design gross weight of about 1600 pounds.

THRUST ENGINE SELECTION

There is only one U.S. turbofan engine which will reasonably satisfy the RSRA requirement, assuming that two auxiliary thrust engines are to be used. The F102-LD-100 engine, manufactured by Lycoming and presently being used in the Northrop A-9 prototype aircraft, delivers an installed thrust of 4100 pounds at 300 KTAS sea level standard day at a specific fuel concumption of .73.

Studies were made to determine the feasibility of using three auxiliary thrust engines to obtain a wider selection. No combination of three engines resulted in a decrease in vehicle weight, and the resulting increase in complexity outweighed any advantage of using three engines (such as cost).

Two F102-LD-100 engines were chosen to provide auxiliary thrust. The engine is physically large (cowled vertical dimension of about 53 inches) which requires that the supporting pylon be inclined upward to diminish engine-wing interference drag. The arrangement of the engine pylon may be seen in Drawing 646-060-200. Removable portions of the thrust engine installation accounts for 10 percent of aircraft design gross weight, so the engine c.g. was placed very near the aircraft c.g. to avoid a significant c.g. shift when the thrust engines were removed.

ROTOR ENGINE REQUIREMENTS

The CH-47C transmission is rated at 3600 SHP at an output of 245 RPM. Allowing for tail rotor, hydraulic and electrical power requirements, and mechanical and installation losses, approximately 2400 SHP (uninstalled) is required from each rotor engine.

A single engine could be used with a considerable saving in weight and complexity, but dual rotor engines are required by the SOW. BHC concurs in this requirement since the RSRA mission includes operation as a pure helicopter, and many test rotors may not be able to perform a satisfactory power-off autorotation.

ROTOR ENGINE SELECTION

The T55-L-7C is an outstanding choice. It's output (2850 SHP) is closer to the desired power than neighboring engines (1900 SHP for the T53-L-702 and 3750 SHP for the T55-L-11A). It has a diameter of 24.2 inches and a length of 44.0 inches. The engine has proven reliability and is available from surplus stocks.

AUXILIARY SYSTEMS

ESCAPE SYSTEMS

The YANKEE extraction system has been selected for the RSRA. BHC is now under contract to the U. S. Navy to study the adaptation of this system to the AH-1J aircraft. It is anticipated that a follow-on contract to design and test the system will be performed. Many of the areas investigated (rotor severance, canopy removal, sequencing, escape path clearances, and option provisions) will be applicable to the RSRA.

The YANKEE system has been in use in the U. S. Air Force T-28 and A-1 aircraft for some time, and enjoys a success rate of over 90 percent. The present effort by BHC is the only known attempt to adapt the YANKEE system to a helicopter. As adapted for helicopter use, the system requires an additional device to prevent extraction in the direction of an obstacle (the earth or another helicopter). The same device (terrain looker) serves as a safety to prevent inadvertant extraction into an unsevered rotor blade in event of a sequencing malfunction.

To afford escape versatility, the crew is offered the options of rotor and/or canopy removal independently of other functions. A delayed parachute opening mode is required for escapes above 250 KTAS.

WING

Wing incidence is variable through a range of ±20 degrees. The wing pivots about two bearings which are attached by inverted "A" frames to the main fuselage beams. The wing balance is housed inside the fuselage carry-through torque box. Lift, drag, and moments all pass through the balance. Lift and drag are reacted through the "A" frame to the fuselage beams, and moments are reacted through the control input arm. See Drawing 646-020-100 for further definition of the wing balance.

The wing sections exterior to the fuselage are separable by means of flange bolts located just inside the fuselage skin. A cover plate is then attached to the fuselage for wing-off operation.

The wing torque box consists of two main spars, constructed of aluminum caps with aluminum-faced honeycomb webs, and top and bottom skins of aluminum-faced honeycomb. The inboard sections are used to carry 1700 pounds of fuel for the 30-minute high-speed cruise mission. All wing fuel is carried inside the torque box.

Split, full-span trailing-edge panels (30-percent chord) are used on both the upper and lower wing surfaces. The upper and lower inboard panels may be operated through a 60-degree range and serve as flaps or drag producers. The outboard lower surfaces operate through a 30-degree range and are used as flaps. The upper outboard panels serve as roll control devices. Drawing 646-020-001 shows wing details.

The roll control panels are controlled by lateral motion of the cyclic stick and powered by a dual hydraulic cylinder. The other trailing-edge surfaces are driven by power hinges through shafts which are interconnected. The lower panels are driven by either of two hydraulic motors, with the other motor operating as a standby. All surfaces (except roll control) may be opened to 30 degrees by the flap position selectors. The inboard surfaces may be opened to 60 degrees by the drag control (speed brake) switch located on the auxiliary engine throttle quadrant. The flap position selectors will not control the inboard surfaces if they are open as a result of a command of the drag control switch and vice versa. The upper inboard panels are driven by a single hydraulic motor. Both the upper inboard and lower surfaces may be closed, if hydraulic failure occurs, by declutching the hydraulic drive motor and allowing airstream pressure to close the surfaces.

The design of power and control systems for the trailing edge surfaces was influenced strongly by their failure requirements. If the upper inboard panel drive power fails, they must be closeable to allow landing with small or inefficient rotors. Should power be lost to the lower surfaces, they must be extendable for the same reason. It was for this reason that dual hydraulic motors were used for the lower flaps.

LANDING GEAR

Because of the requirement that the aircraft be operable without a wing, inclusion of the main landing gear in the wing was precluded. Even if this requirement did not exist, the requirement to vary wing incidence, measure wing beamwise and chordwise forces, and allow high landing sink rates effectively precluded placing the gear in the wing.

The first main landing gear design involved placing the gear trunnions in fuselage stub-outs which fit the wing's contour, but were a part of the fuselage, and did not tilt with the wing. The wheels were retracted directly inboard. This scheme was discarded when the decision was made to provide large wing incidence change, since a discontinuity exists at the junction of the stub-out and the tilting portion of the wing, when the wing tilts through large angles.

The only alternative was to place the main landing gear in the fuselage, but for a conventional air/oil oleo strut, it was impossible to obtain adequate tread. If the strut were canted outboard to gain tread, unacceptable tire scrubbing occurred in helicopter landings, since vertical motion of the wheel also meant horizontal motion. The main landing gear pictured in Drawing 646-050-001 (Sheet 1) solved this problem by allowing purely vertical motion of the wheel. This gear is very similar to that used in the BHC Model 300 proprotor aircraft.

The tail wheel is of conventional design and has a stroke equalling that of the main landing gear, to avoid pitching problems. The tail wheel is partially covered by the ventral fin, the lower portion of which turns with the tail wheel for steering. Drawing 646-050-001 (Sheet 2) depicts the tail wheel installation.

ROTORS

The cost of developing and manufacturing a prototype rotor historically exceeds one million dollars. To avoid this expenditure, existing BHC rotor designs were surveyed to determine if they were suitable for the RSRA. The BHC Model 240 main and tail rotors were found to be suitable. An analysis was made of the Model 240 main rotor to determine vibration characteristics throughout the RPM and flight speed range anticipated for the RSRA. See the BHC RSRA Conceptual Study Report for a discussion of this analysis. The Model 240 rotors are defined in the "Aircraft Description" section of this report.

It was determined that the necessary alterations to the Model 240 main rotor would include a radius extension of 0.5 foot, a decrease in blade twist from minus eight to zero degrees, and changing of blade weights. All blade section properties remain unchanged. It will be necessary to provide a new blade tool, but this can be done for approximately one-tenth the price of obtaining a new blade design.

The Model 240 main rotor blade, altered as described above, will allow flight to a speed of 300 KTAS, provided that the blade is allowed to operate at a tip-Mach number of one.

VARIABLE GEOMETRY ROTOR

The Variable Geometry Rotor was designed to allow variations in hub azimuth and vertical hub spacing for a dual-hubbed, 6-bladed, hingeless rotor. Vertical spacing of the flexure centerline may be varied between 2 and 10 inches. Azimuth spacing may be selected in 4-degree increments.

Both azimuth and vertical spacing is achieved with the upper hub (refer to Drawing 646-010-100). Azimuth spacing is achieved by loosening the flexure retention bolts sufficiently to allow the flexure indexing teeth to clear, then manually rotating the upper flexure to the desired position and retorqueing the flexure retention bolts. A change in vertical spacing is almost as simple: the upper hub cone bolts are loosened, and the threaded insert locking bolts are removed; the threaded insert is then rotated until the desired spacing is achieved, and the bolts reinstalled and retorqued. One revolution of the insert will cause a change in vertical spacing of 0.5 inch. The pitch links must be adjusted each time the upper hub is moved, except for small motions.

Azimuth changes will require about 15 minutes. Hub vertical adjustment will require about 20 minutes. No tools are required, other than hand wrenches.

DRIVE TRAIN

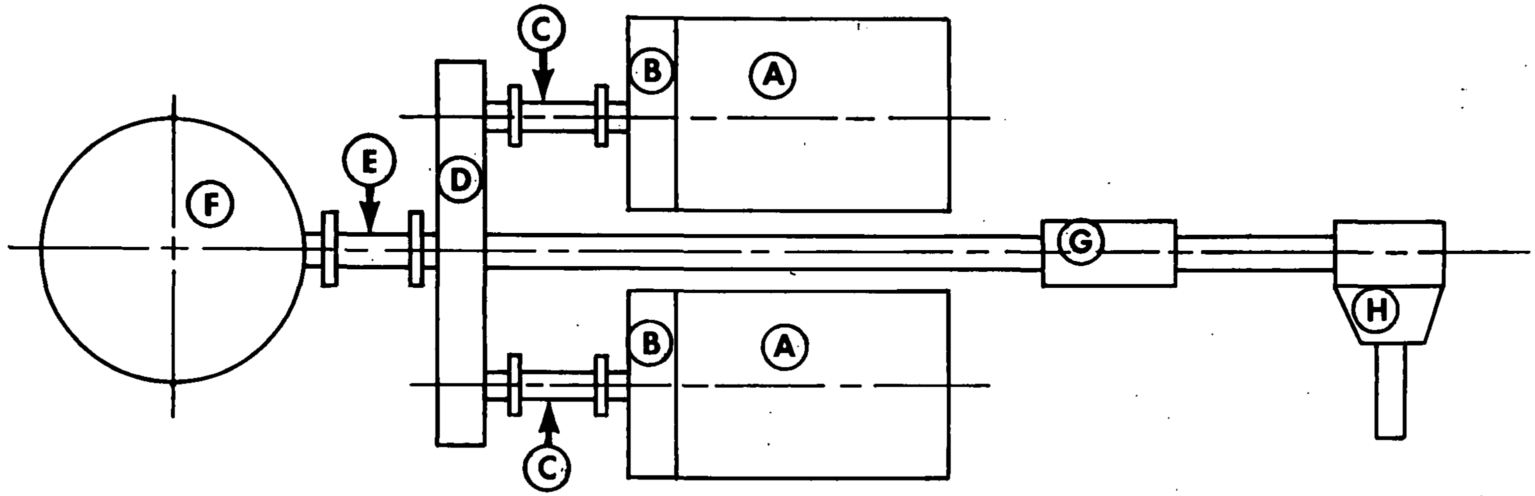
The RSRA drive train is composed of those components defined in Figure 8. The speed decreasers (existing BHC hardware) mount directly to the face of the T55-L-7C engines, providing a reduction of 2.0306:1, allowing an output at 6300 RPM. The driveshaft assembly (speed decreasers to combining gearbox) is also off-the-shelf BHC hardware. The combining gearbox distributes the power of the two main rotor engines to the main and tail rotors, hydraulic pumps, and electrical generators. A free-wheeling unit is used on each input shaft.

The combining gearbox is attached to the airframe by four elastomeric mounts. Helical gears are used exclusively for quietness. The combining gearbox includes a pressure pump, oil cooler, oil filter, electric chip detector, and associated hardware. It is shown in Drawing 646-040-200.

The combining gearbox is coupled to the main rotor transmission by a driveshaft which allows pylon motion through spherical splines at both ends of the shaft. The main rotor transmission is the forward transmission from a Boeing Vertol CH-47C. This gearbox is rated at 3600 SHP at 245 output RPM.

The intermediate gearbox, tail rotor gearbox, and connecting driveshaft are outgrowths of the BHC UTTAS Program. The tail rotor drive train is rated at 410 SHP with an overload ability of 550 SHP.

Tilting of the main rotor shaft is accomplished as a ground operation. The input shaft to the CH-47C transmission droops 8 degrees. Each of the two couplings between the combining



Letter	Component	Manufacturer	Used On
A	Turbine	AVCO-Lycoming	Bell Model 309 "KingCobra"
B	Speed Decreaser	Bell	Bell Model 309 "KingCobra" Bell Model 211 Test Vehicle
C	Driveshaft Assembly	Bell	Bell Model 309 "King Cobra"
D	Combining Gearbox	Bell	New
E	Driveshaft Assembly	Bell	New
F	Main Rotor Gearbox	Boeing Vertol	CH-47C
G	Intermediate Gearbox	Bell	Bell Model 240
H	Tail Rotor Gearbox	Bell	Bell Model 240

Figure 8. RSRA Drive Train.

gearbox and the main rotor gearbox is capable of two degrees of misalignment for steady-state operation, which allows the transmission to be inclined as far aft as four degrees forward. Forward tilt of the transmission, up to 12 degrees, is then accommodated by the two couplings. If the Servo-Null vertical vibration attenuation system is used, tilt of the transmission is accomplished by differentially moving the fore and aft focused pylon attachment arms (see Drawing 646-010-400). If only a focused pylon is used for vibration attenuation, transmission tilt is accomplished by changing the length of the fore and aft restraint links.

VIBRATION ATTENUATION SYSTEM

The vibration attenuation problem may be divided into two categories: vertical vibrations and inplane (horizontal) vibrations. Inplane vibration attenuation will be accomplished by the BHC focused pylon system, regardless of the method chosen for vertical vibration attenuation. The focused pylon principle has been proven to be a reliable means of attenuating inplane oscillatory forces.

The desirability of including a vertical vibration attenuation system as a part of the RSRA program is questionable. Until the response characteristics of the RSRA fuselage are predictable, an exact definition of the vertical vibration attenuation system is not practical. It may well be that the RSRA rotor test ability, without vertical vibration attenuation, is great enough that such a system would not be required except for a few rotor types and sizes.

In the event that a vertical vibration attenuation system is required, two principles presently under study by BHC appear to offer practical solutions.

The Servo-Null System is a device which has a natural frequency below that of probable test rotors. Such a system is very soft and requires a servo-controlled hydraulic-nulling device to maintain pylon vertical position with varying vertical loads. Because of its low natural frequency, it is capable of attenuating vertical forces over a wide frequency range. This device is shown in Drawing 646-010-400. A sketch is included as Figure 9.

The Nodalized Beam System has been in flight test at BHC for some time, with very favorable results. This device has a natural frequency near that of the most significant rotor mode. It has a much more narrow frequency range than does the Servo-Null System, but this range can be adequate for the RSRA flight speed and rotor RPM envelope if the blade tips are allowed to operate at higher Mach numbers (up to 1.0).

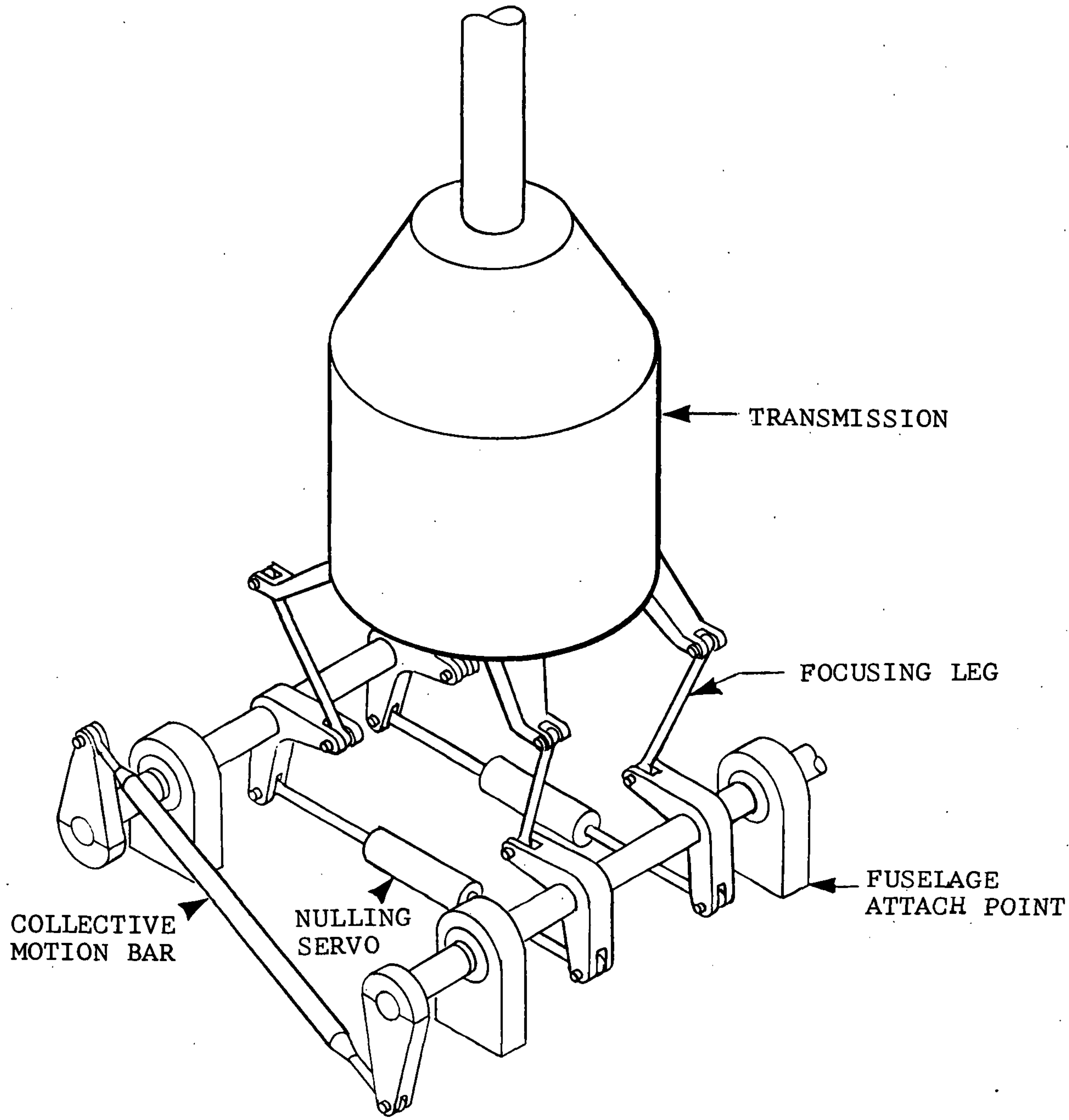


Figure 9. The Servo-Null System.

ELECTRICAL SYSTEM

The AC electrical system is powered by a single 15 KVA generator, driven by a constant-speed drive, which mounts to a pad on the combining gearbox. The constant-speed drive and generator are oil cooled and require an external 25 HP cooler. A 750 VA standby inverter is also included in the AC system. The total AC load is approximately 11,100 VA.

The DC system is powered by two 300-ampere starter-generators and a 34-ampere hour battery. The starter-generators are derated to 200 amperes for generating. Two starter-generators are mounted on the auxiliary engines, but used only for starting. The DC load is approximately 100 amperes.

HYDRAULIC SYSTEM

The hydraulic system uses dual pumps and dual hydraulic actuators. The system operates at 3000 psi and is designed in accordance with MIL-H-5440.

One of the dual hydraulic systems powers the SCAS and force feel actuators, in addition to powering one side of the dual flight control servoactuators.

The second hydraulic system powers all utility functions (landing gear, brakes, flaps) in addition to powering the other side of the dual flight control servo-actuators.

All utility hydraulic circuits are automatically isolated from the second hydraulic system in event of fluid loss from the utility system.

COCKPIT ARRANGEMENT

GENERAL

The cockpits have been arranged to accomplish the following:

- Provide the pilot with control of all aircraft functions.
- Provide the pilot with a complete instrument panel.
- Provide the copilot/computer operator with adequate controls and instruments to safely land the vehicle under any flight conditions.
- Provide the third crewmember with cyclic rotor and fixed-wing controls.

REAR COCKPIT

The rear cockpit accommodates the pilot and third crewmember. All instruments and control switches are located as near the pilot's station as practical. All flight instruments are located about the centerline of the pilot's forward vision. Engine instruments are located in order of engine locations on the aircraft. A switch for control of the main rotor longitudinal cyclic washout actuator is mounted on the pilot's cyclic stick. Collective washout of the main and tail rotors is by means of switches mounted on the collective stick. The positions of the washout actuators are presented on the instrument panel.

The third crewmember has a cyclic stick and directional control pedals, but no washout controls, collective stick, or brakes.

Considerable space remains on the rear instrument panel for additional presentation. Drawings 646-070-200, Sheets 1 and 2, depict the rear cockpit.

FRONT COCKPIT

The copilot's position has adequate instrumentation and controls to allow flying of the aircraft in any weather conditions. Initial pilot checkouts would also be accomplished from the copilot's station. Control of the lower flaps and wing incidence is afforded, but control of the upper flaps (or both flaps when extended as speed brakes) is limited to emergency retraction. Drawing 646-070-300 shows the copilot's cockpit.

CONTROL SYSTEMS AND STABILITY AND CONTROL

ELECTRONIC CONTROL SYSTEM

The total computer operated control system is shown in Drawing 646-070-100. This drawing shows each subsystem's relation to the total control system. Each channel (fore and aft, lateral, etc.) is shown as an independent system; they are connected to other channels only through the aerodynamic response and computer control loop. All commands from the computer are dual. The drawing shows two lines from the computer to each control system. This represents two complete sets of signals, two for fly-by-wire (FBW) commands, two for rotor control, two for gain change, etc. Each independent channel (fore and aft, lateral, flaps, throttle, etc.) is shown in detail in block diagram form on other figures.

Cyclic Control

The longitudinal cyclic control system is shown in Drawing 646-074-200. The lateral cyclic control system is identical to the longitudinal control system. Actuators 4 and 5 are dual fuselage SCAS actuators which provide the desired fuselage stability and shape the pilot inputs to provide the desired response. These are dual to provide large control authority, but small failure effects. If a single actuator were used, and limited to ±10 percent authority, the maximum failure input would be 20 percent. With a dual system, however, each could have ±10 percent authority or a total of ±20 percent authority with failure inputs limited to around 5 percent or less. The reason is that if one of the dual systems fails and produces a bad input, the other will subtract out most of the bad input and automatically shut off both when such a disagreement exists. Upon shutoff, each actuator will slowly return to its center position and lock.

Actuator 1 is used to provide autopilot or computer-controlled commands to the control system. During these modes the pilot's hand is off the stick, allowing the controls to be positioned by Actuator 1. During manual control by the pilot, Actuator 1 is used to provide the desired stick control forces and trim. Actuator 1 system is monitored continuously for failures. If an electronic failure occurs, the valve hydraulic flow is restricted to prevent stick jump. If a hydraulic hardover failure occurs, the actuator will bypass within 0.5 inch of stick travel.

Rotor Control Actuators 2 and 3 provide the independent rotor control required for this system. This dual rotor control system receives its commands from transducers which measure rotor parameters such as flapping, control moments, pylon displacements, rotor lift, etc., and makes the required motion to control the parameter of interest. The total motion of these actuators must be the difference between the pilot input to the rotor and the total rotor control motion required. If a particular requirement were to hold rotor controls

fixed at a certain position, then Actuators 2 and 3 would provide motion to subtract out all pilot inputs in order to have no rotor change. These two actuators will share the motion requirements equally. Rotor control actuators are dual for safety reasons. These actuators must have a large control authority (about ± 20 percent each) in order to control the rotor independently for a wide range of flight conditions. A hardover failure of either actuator will cause the good actuator to go to the opposite direction to take the bad input out. At the time when the displacement difference reaches the "fail" level (about 5 percent), both actuators will be disabled and returned slowly to center and locked. Pilot inputs will now go directly to rotor controls as well as to the fuselage controls. The input to the rotor produced by the failure will not be the 5 percent difference in actuator position, but something less which is the difference between the "bad" actuator input and the "good" actuator cancellation of the bad input.

The gain change mechanism (Figure 10) is used to control the gain between the control system and the inputs to the rotor. A range of zero to 100 percent is required. The gain is set to zero to lock out the rotor controls. This device provides the ability of programmable or manual gain control, either with or without the independent rotor control system. Typical functions might be to reduce rotor control sensitivity with increasing airspeed, to limit maximum rotor inputs, provide different control gain for different rotors, manually control rotor lockout, etc. The gain change mechanism is placed downstream of the independent rotor control actuators in order to change the gain of that system as well as the mechanical system. Not only can controls be locked out but all failures upstream can also be locked out of the rotor. The gain change mechanism is a low velocity system which is continuously monitored for failures. Should a failure be detected, the gain change mechanism actuator will be shut off and locked in that position. A failure of the gain change mechanism control system does not create dangerous conditions since the pilot can put mechanical inputs in to override the actuator and the aircraft can be flown and landed with this device locked in the position in which it failed.

Yaw Control

The yaw control system is similar to the longitudinal and lateral control systems. If later studies show that the safety and performance requirements can be met without the dual actuators shown, a single actuator system will replace the dual system.

Main Rotor Collective

Main rotor collective controls are identical to the main rotor cyclic portion of the longitudinal and lateral control systems including Actuators 1, 2, 3, and the gain change mechanism. The function of each portion of the main rotor collective system is the same as it is for the main rotor cyclic systems.

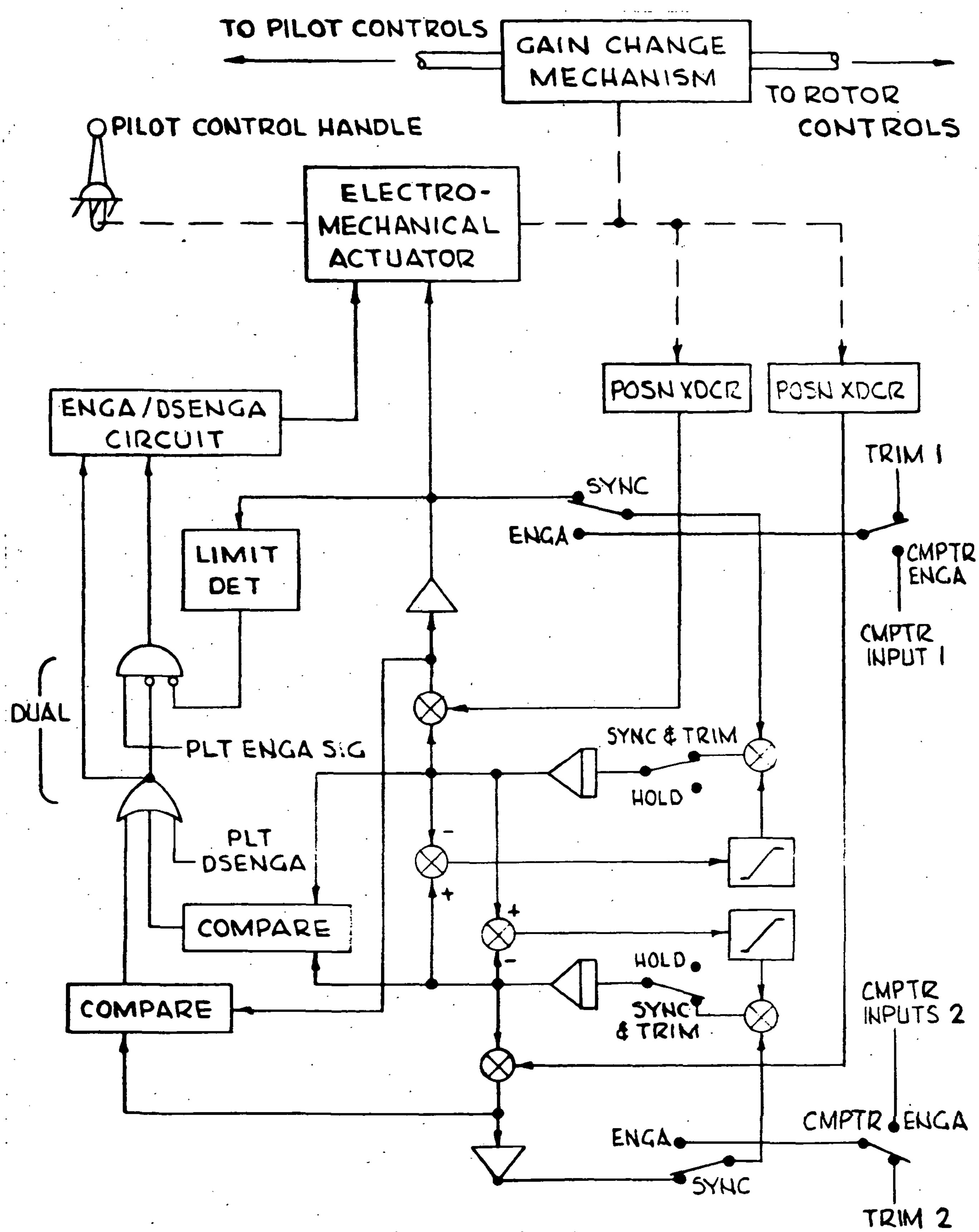


Figure 10. Gain Change Mechanism Control System.

Throttle

Figure 11 shows the throttle control system. The speed of operation of the throttle actuators is dictated by the requirement for rapid thrust reduction when failure of both rotor engines occurs at high flight speeds. Such rapid throttle manipulation is not required for speed holding functions, and the actuator hydraulic fluid flow is restricted to allow motion rates of 5-10 percent per second in this mode. During manual throttle operation, the actuators are bypassed. During automatic throttle control, the control handles follow throttle movements. Automatic system control will be shut off whenever a "fail" tolerance of about five percent is exceeded. The pilot can override the automatic system at anytime, which will result in disengagement of the overridden system.

Flaps

Lower and upper flap control systems detail block diagrams are shown in Figures 12 and 13 respectively. Lower flaps are controlled by redundant electronically-controlled hydraulic motors. The engage/disengage logic shown in the figure is such that System 1 will be the primary system and will control the lower flap with System 2 operating as monitor. While performing monitor functions, System 2 is operating with the hydraulic pressure off. A loss of hydraulic pressure by System 1 will result in engaging System 2 to operate the lower flaps. An out-of-tolerance disagreement between electronic signals in System 1 and System 2 will automatically disengage both systems. A pilot emergency override switch will be provided which will bypass all the system electronics in Systems 1 and 2 and will operate the lower flaps from whichever system has electrical and hydraulic power.

The upper flaps operate like the lower but without redundant hydraulic motors. After an electronic failure which causes automatic disengagement, the upper flaps may be operated by the pilot override switch if electrical and hydraulic power is available. A pilot emergency release is also provided to declutch upper flaps to allow them to close after a loss of hydraulic pressure.

Speed Brakes

Upper and lower inboard flaps operate as speed brakes by applying equal signals to both. This may be a pilot- or computer-controlled function.

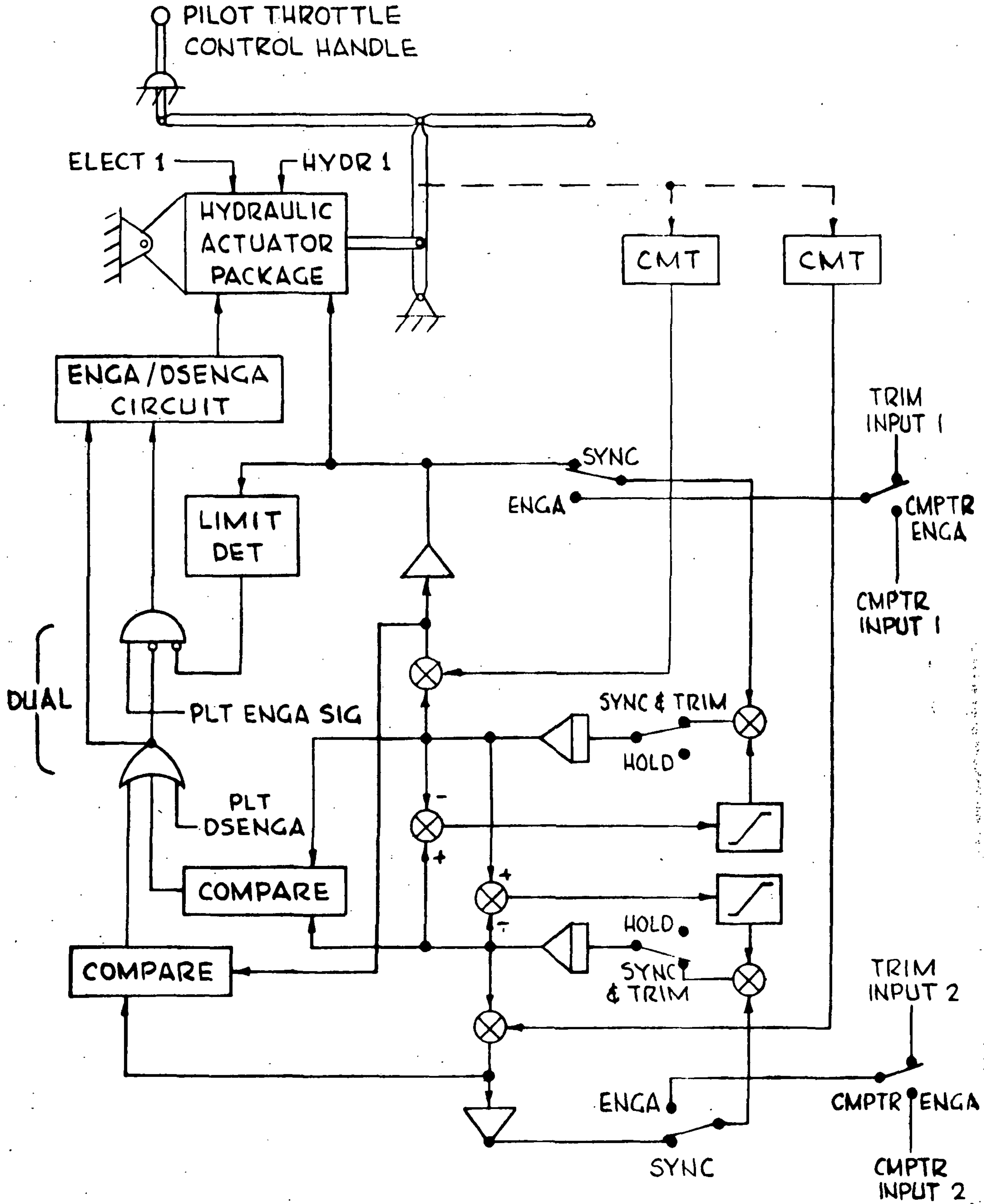


Figure 11. Throttle Control System.

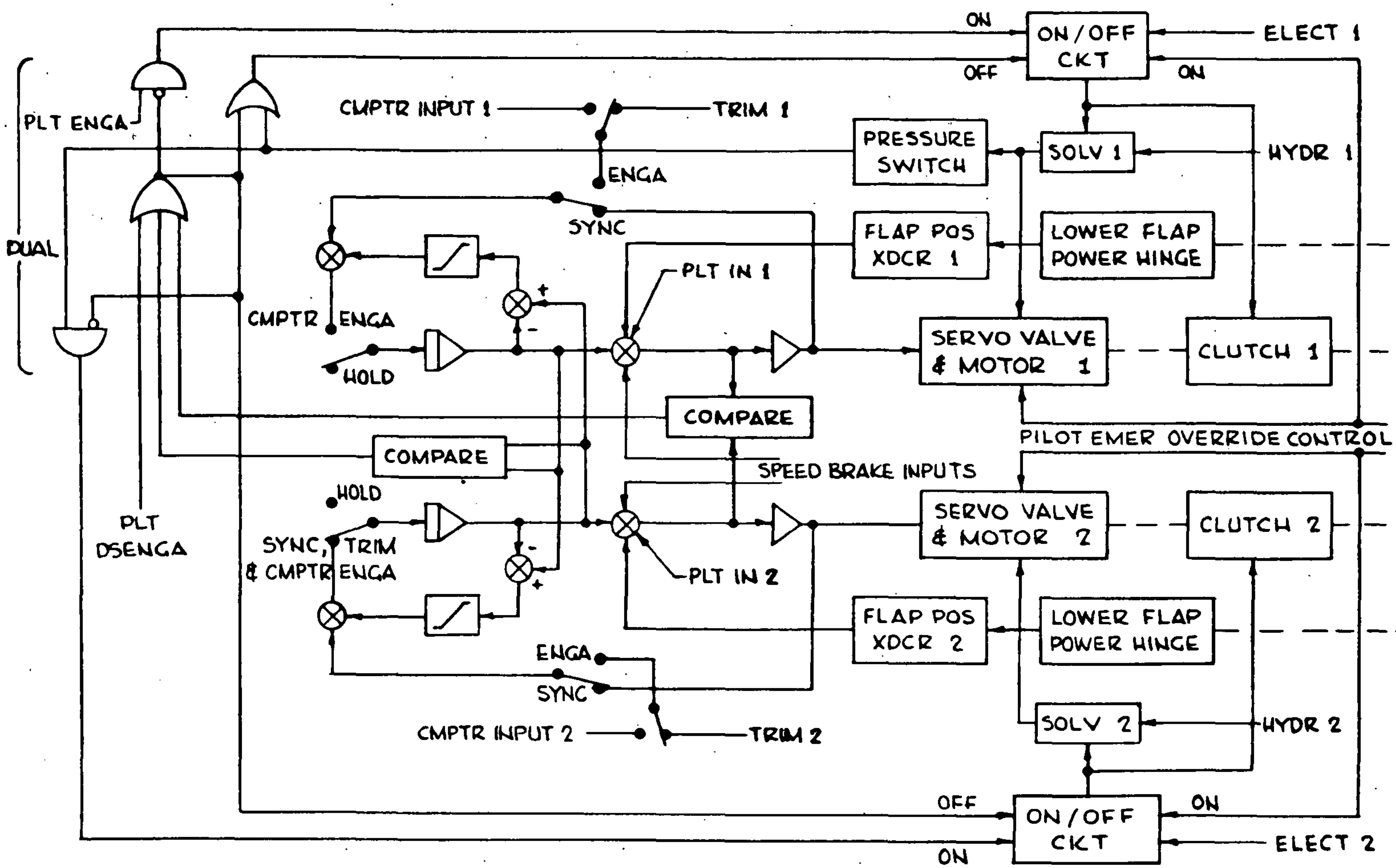


Figure 12. Lower Flap Control.

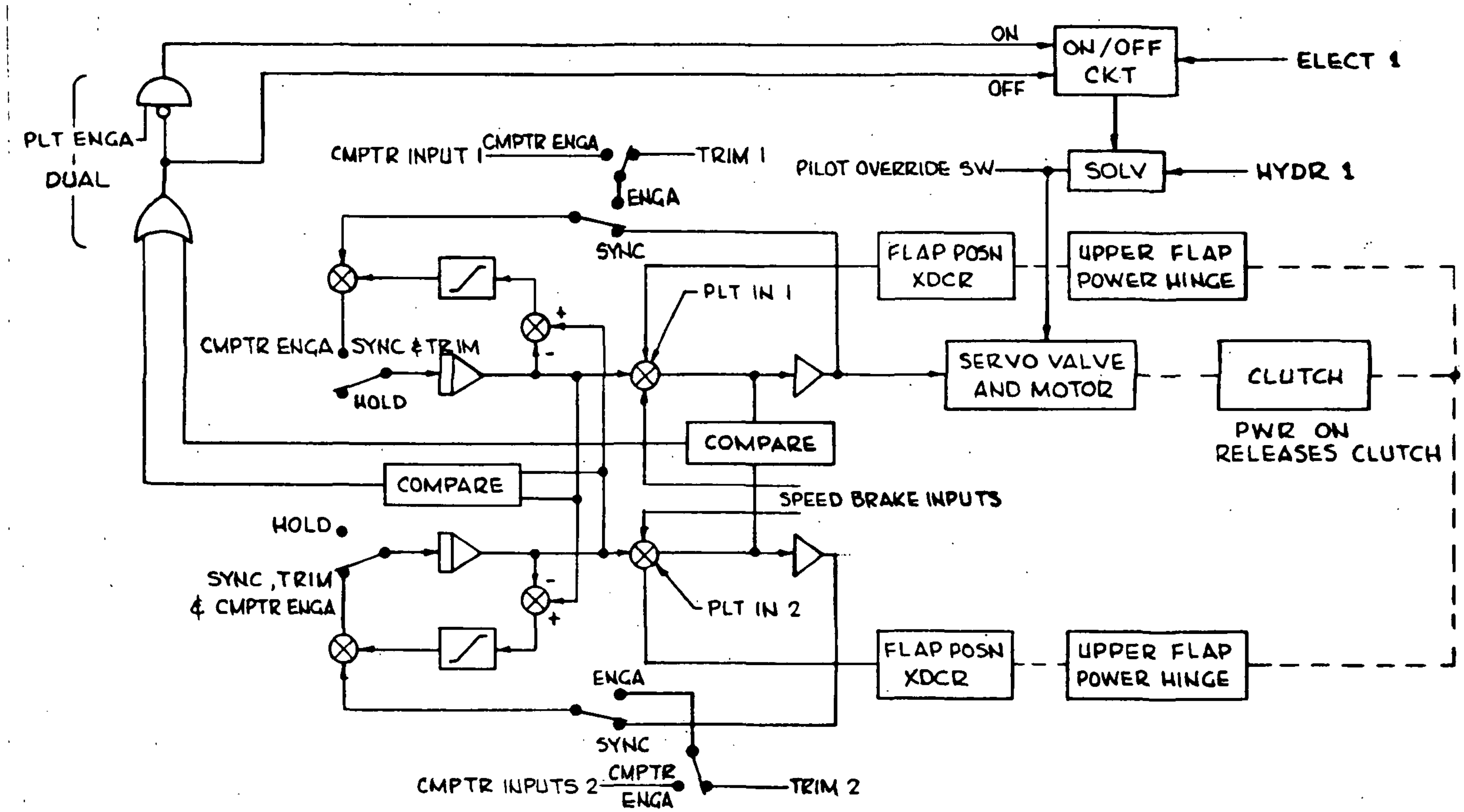


Figure 13. Upper Flap Control.

COMPUTER - CONTROL SYSTEM INTERFACE

Guidelines and Philosophy

The RSRA electronic flight control system (EFCS) has been designed to interface with a digital computer which will be specified at a later date. Some of the interface requirements and interface philosophy are discussed in subsequent paragraphs. During detail design of the EFCS and computer, these requirements may change as some functions currently incorporated within the EFCS could be accomplished within the computer, or vice versa. The basic guidelines which were used in defining the overall computer/EFCS interface are:

- The computer/EFCS system shall be fail safe.
- All flights may be aborted after any single failure within the computer or EFCS.
- Normal engagement and disengagement of computer-controlled modes shall not result in sudden transients.
- All pilot-operated controls (stick, pedals, throttles, etc.) shall remain in the position existing at the time of disengagement of a computer mode.
- After disengagement of a computer mode, the series actuators should return to neutral at a rate which can be easily compensated by the pilot. A washout time constant on the order of five seconds is reasonable.

In order to meet these guidelines, the following computer/EFCS interface philosophy was established:

- All computer inputs to the EFCS must be dual. These inputs are switched in or out within the EFCS when the computer control mode is engaged or disengaged.
- All computer command signals must be zero at the time of engagement.
- All EFCS channels will incorporate an integration and holding circuit in the computer control signal path to provide a servo transfer function of the following form:

$$\frac{O}{i_c} = \left(K_1 + \frac{K_2}{s}\right)\left(\frac{1}{\tau s + 1}\right)$$

O = servo output

i_c = computer command

τ = servo response time - sec

s = frequency - rad/sec

The factor K_2 will be selected to be compatible with the overall computer/EFCS/aircraft dynamics. This function will result in zero steady-state error command signals from the computer, thus minimizing disengage transients. The series actuator channels will incorporate a signal washout function to return the actuator to neutral following disengagement.

- The EFCS channels are redundant with monitoring and disengage circuits to detect failures within the EFCS. Dual, independent transducers provide redundant signals for the computer. In order to retain the overall safety and failure philosophy, the computer must provide independent dual signals for the redundant control channels. The computer must be redundant, <u>or</u> it must contain internal monitoring to the level necessary to detect any failure within the computer.

- Engage/disengage control signals to the computer will be interlocked with the appropriate EFCS channels so that computer engagement cannot occur unless the EFCS is functioning properly.

- Engage logic signals from the computer to the appropriate EFCS channels will be present when the computer monitoring functions indicate that the computer is functioning properly. (This logic channel is not required if independent dual computers are used.)

Input Signals to the Computer

The EFCS will input the following signals to the computer:

Signal	Range	Format	Scale	Remarks
Rotor Flapping	±15°	AC Analog	±10V (rms)	Dual
Mast Tilt	±20°	AC Analog	±10V (rms)	
Rotor Shaft	±360°	AC Resolver		
Fuselage Angle of Attack	±30°	DC Analog	±10V	
Rotor Moments	± Max	DC Analog	±10V	
Rotor Lift	± Max	DC Analog	±10V	
Rotor Drag	± Max	DC Analog	±10V	
Fuselage Pitch	±70°	3-wire Synchro		
Fuselage Roll	±70°	3-wire Synchro		
Roll Rate	±1 rad/sec	AC Analog	±10V	
Pitch Rate	±0.5 rad/sec	AC Analog	±10V	
Yaw Rate	±0.5 rad/sec	AC Analog	±10V (rms)	
Rolling Acceleration	±1.0 rad/sec^2	DC Analog	±10V	
Pitching Acceleration	±0.5 rad/sec^2	DC Analog	±10V	
Yawing Acceleration	±0.5 rad/sec^2	DC Analog	±10V	
Directional	±360°	3-wire Synchro		
Sideslip Angle	±60°	DC Analog	±10V	
Normal Acceleration	±3 g's	DC Analog	±10V	
L Aux Thrust	±100%	DC Analog	±10V	
R Aux Thrust	+100%	DC Analog	+10V	
Airspeed	+300 kt	DC Analog	+10V	
Stick Motions:				
F/A	0-100%	DC Analog	±10V	
Lateral	0-100%	DC Analog	±10V	
Directional	0-100%	DC Analog	±10V	
Collective	0-100%	DC Analog	±10V	
Flap Position:				
Lower	-60°	DC Analog	±10V	
Upper	+30°	DC Analog	±10V	Dual

Computer Outputs

The EFCS will accept the following computer outputs:

1. Dual Analog Outputs (+10V Full Scale)

   F/A Rotor Control (No. 1 and No. 2)
   F/A Parallel (No. 1 and No. 2)
   Lateral Rotor Control (No. 1 and No. 2)
   Lateral Parallel FBW (No. 1 and No. 2)
   Directional Parallel FBW (No. 1 and No. 2)
   Collective Rotor Control (No. 1 and No. 2)
   Collective Parallel FBW (No. 1 and No. 2)
   F/A Rotor Gain Change Mech (No. 1 and No. 2)
   Lateral Rotor Gain Change Mech (No. 1 and No. 2)
   Collective Gain Change Mech (No. 1 and No. 2)
   Directional Gain Change Mech (No. 1 and No. 2)
   Lower Flap Control (No. 1 and No. 2)
   Upper Flap Control (No. 1 and No. 2)
   Left Auxiliary Engine Control (No. 1 and No. 2)
   Right Auxiliary Engine Control (No. 1 and No. 2)

2. Logic Outputs

   Speed Brake - Flap Position Logic (No. 1 and No. 2)

   An engage logic signal for each pair of analog control signals listed above. (This signal shall be present if the computer is engaged and is functioning, and if all signal inputs and computations associated with the control output signal are "good". Loss of this signal will cause the EFCS to automatically disengage computer-controlled modes.)

Failure Effects (Single Failure)

1. Electrical Power Failure

   Loss of electrical power will shut off a solenoid valve to remove hydraulic power from the system. Series actuators will slowly center and lock and parallel actuators will by-pass to allow manual operation. The pilot will be notified by the master caution panel.

2. Hydraulic Power Failure

   Loss of hydraulic power will cause the same results as loss of electrical power.

3. <u>Electronic Failure</u>

Failure in either of the dual electronic channels of any control system will result in both channels being deactivated. This is done by the constant comparison of both outputs, which automatically disengages both when disagreement occurs. Series actuators will slowly center and lock while parallel actuators bypass.

4. <u>Hydraulic Failures</u>

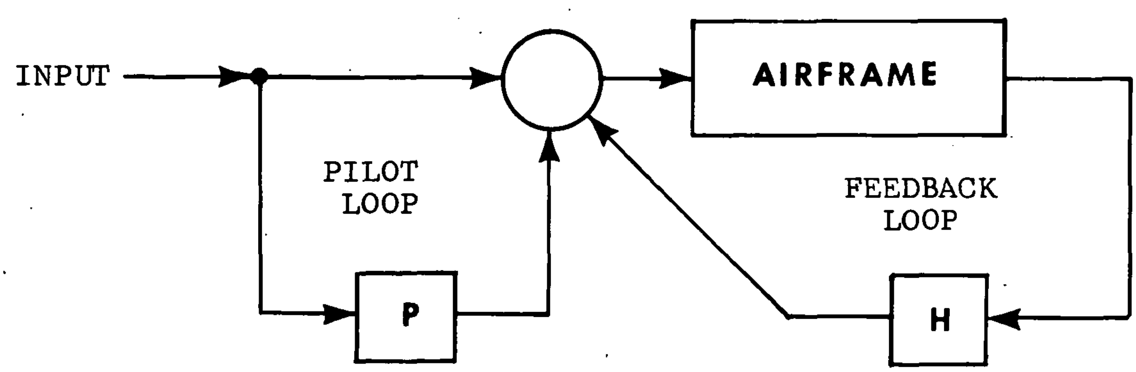
A hydraulic failure in the control system will have the same results as an electronic failure.

STABILITY AND CONTROL AUGMENTATION SYSTEM

The Stability and Control Augmentation System (SCAS) has two basic modes of operation--rotor control and fixed-wing control. The rotor control SCAS is primarily intended for low-speed operation up to 150 knots, although it is effective up to 200 knots. The fixed-wing SCAS is intended for the higher speeds. During detail design, the optimum change-over speed will be determined. It may be desirable to phase the two modes in and out over a speed range -- say 150 to 200 knots. All channels of both the rotor control SCAS and fixed-wing SCAS are dual with failure detection circuitry. System diagrams of the SCAS and related EFCS components are shown in Drawing 646-074-200.

Root locus plots which depict the effect of the SCAS on longitudinal and lateral stability are shown in Figures 14 and 15 for the compound configuration and Figures 16 and 17 for the pure helicopter configuration.

Table VIII presents the SCAS transfer functions and applicable numerical values of equation components. The transfer functions for "H" and "P" refer to the Feedback and Pilot Loops, respectively, as shown in the following sketch:



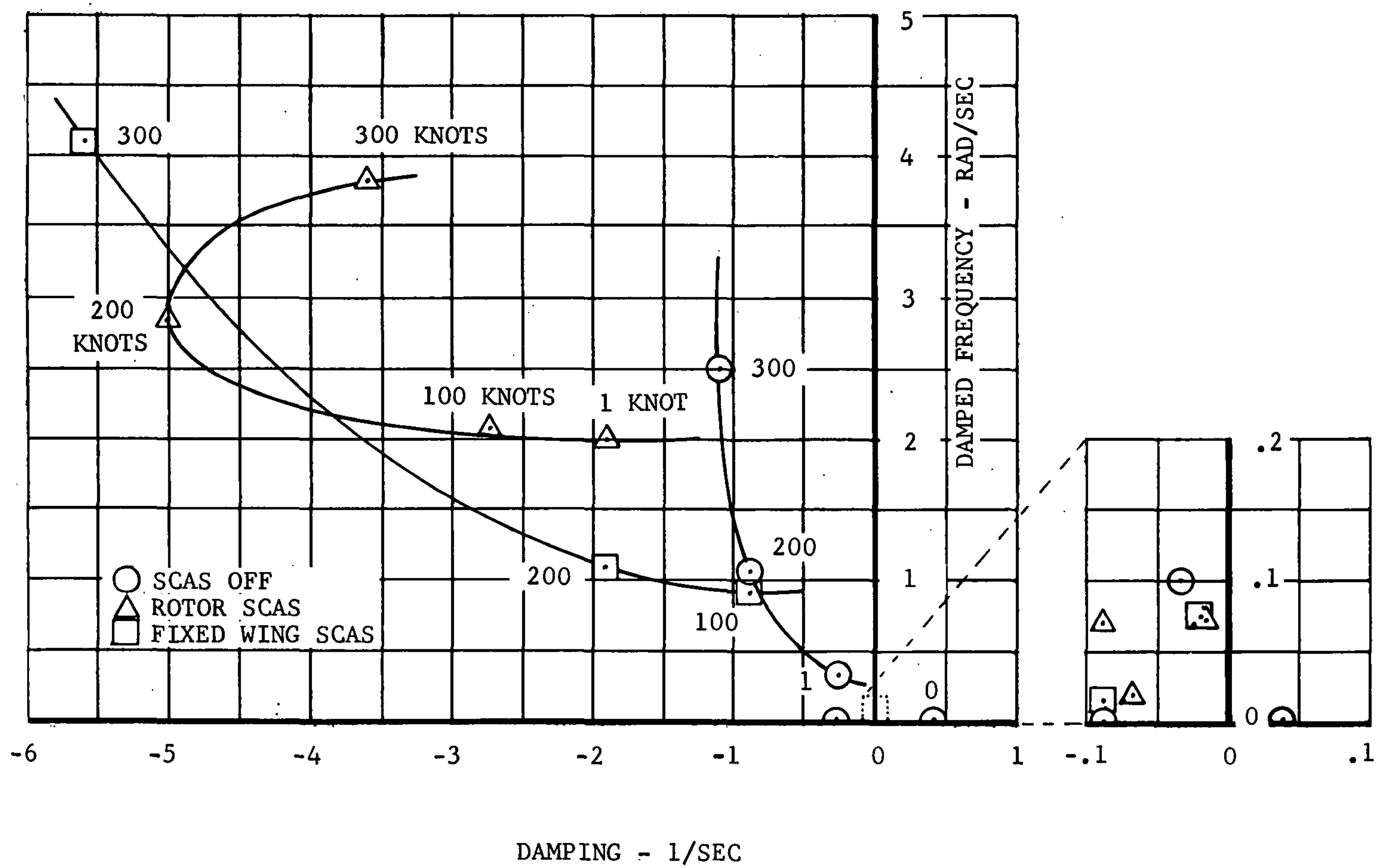


Figure 14. Effect of SCAS on Longitudinal Stability, Compound Configuration.

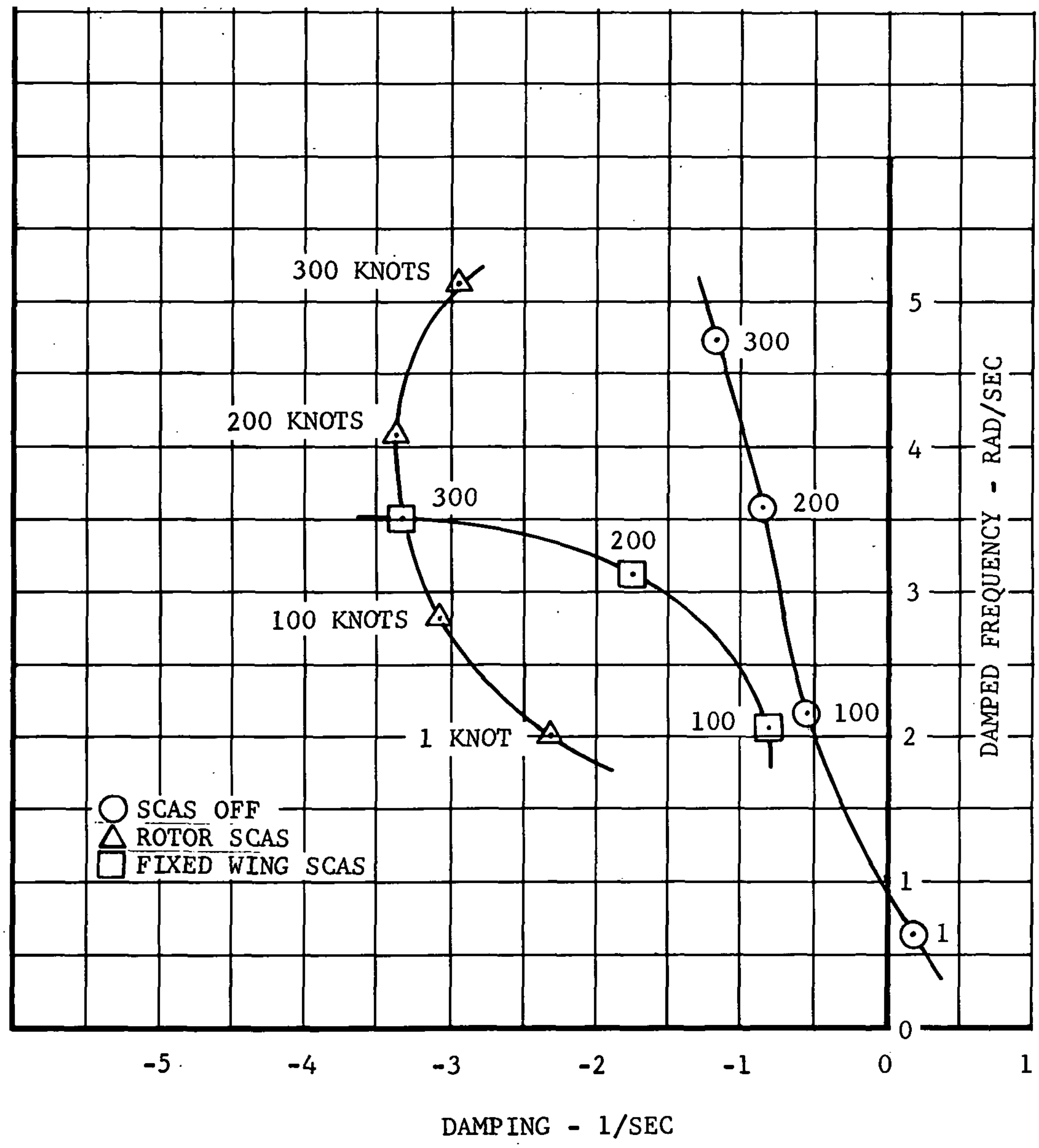


Figure 15. Effect of SCAS on Lateral Stability, Compound Configuration.

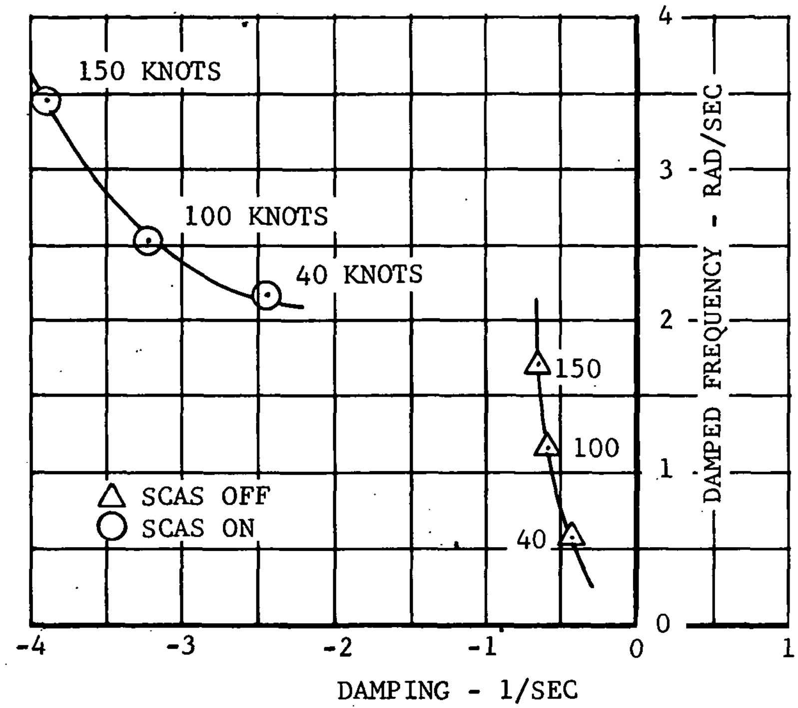


Figure 16. Effect of SCAS on Longitudinal Stability, Helicopter Configuration.

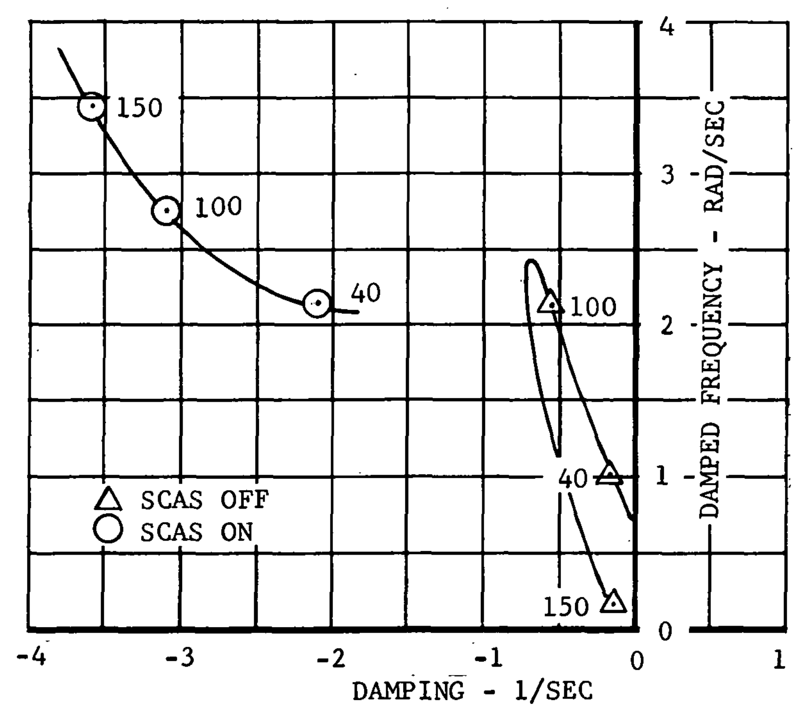


Figure 17. Effect of SCAS on Lateral Stability, Helicopter Configuration.

TABLE VIII. SCAS TRANSFER FUNCTIONS

	KFB	T1	T2	T3	T4	T5	T6	KP	T7
Pitch (Rotor)	.333	18	1	.5	3	3	.08	35	0
Roll (Rotor)	.14	16.5	1	.67	3.33	3.33	.08	33.3	0
Yaw (Rotor)	.07	15	1	.42	2.5	2.5	.05	14	.4
Pitch (Elevator)	.5	16	1	1	4	4	.08	30	1
Roll (Ailerons)	.02	0	1	0	0	0	0	0	0
Yaw (Rudder)	.04	0	1	0	0	0	0	0	0

$$H = \frac{KFB\ S\ (T_1\ S + T_2)\ (T_3\ S + 1)}{(T_4\ S + L)(T_5\ S + 1)(T_6\ S + 1)}$$

$$P = \frac{KP\ S}{(T_7\ S + 1)(T_4\ S + 1)(T_5\ S + 1)(T_6\ S + 1)}$$

Table IX presents transfer functions for the rotor controls at 1 knot, 200 knots, and 300 knots. These transfer functions are for the compound mode, with fixed-wing control surfaces immovable.

Table X presents transfer functions for the fixed-wing controls at 200 and 300 knots, with the rotor controls fixed.

Table XI presents combined transfer functions for the compound mode at 200 knots, with rotor controls operable, but SCAS operable for only the fixed-wing controls.

These analyses were accomplished using rigid-body equations. Some preliminary analysis was performed with equations which incorporate rotor and pylon dynamics, which indicated that some stability problems may exist because of interaction between the SCAS and pylon modes. This must be examined in detail in the design program, as soon as pylon dynamics can be predicted accurately.

Rotor Transfer Functions

Open-loop rotor response to rotor control inputs at 200 knots is given in Table XII. In any closed-loop control of the rotor, through the digital computer (such as when conducting rotor flight tests), the rotor and pylon dynamics must be considered, but these functions may be used for preliminary design. The final design of the computer, EFCS, and SCAS must consider these components as an integrated system.

TABLE IX. ROTOR TRANSFER FUNCTIONS

Axis	Control	Speed (kts)	Transfer Function, SCAS Off (deg/in.)	Transfer Function, SCAS On (deg/in.)
Pitch	Main Rotor (fore & aft)	1	$\frac{8.3(S+.0035)(S+.3465)}{(S-.4)(S+.28)(S+.25\pm j.36)}$	$\frac{8.3(S+.0035)(S+.025)(S+.35)(S+6.6\pm j3.7)}{(S-.11)(S+.35)(S+.08\pm j.066)(S+1.9\pm j2.0)(S+9.4)}$
Pitch	Main Rotor (fore & aft)	200	$\frac{12.6(S+.04)(S+.829}{(S-.04)(S+.078)(S+.807\pm j1.1)}$	$\frac{12.6(S+.025)(S+.04)(S+.83)(S+6.6\pm j3.7)}{(S-.017)(S+.066\pm j.02)(S+.77)(S+3.9)(S+5.0\pm j2.8)}$
Pitch	Main Rotor (fore & aft)	300	$\frac{12.4(S+.058)(S+1.25)}{(S+.033\pm j.103)(S+1.15\pm j2.5)}$	$\frac{12.4(S+.025)(S+.058)(S+1.25)(S+6.6\pm j3.7)}{(S+.13)(S+.83)(S+.017\pm j.073)(S+3.6\pm j3.8)(S+7.4)}$
Roll	Main Rotor (Lateral)	1	$\frac{21.7(S+.05)(S+.76)}{(S-.32)(S+.85)(S-.17\pm j.62)}$	$\frac{21.7(S+.025)(S+.053)(S+.76)(S+1.01\pm j2.43)(S+13)}{(S-.11)(S+.79)(S+.127\pm j.097)(S+2)(S+1.66\pm j1.88)(S+9)}$
Roll	Main Rotor (lateral)	200	$\frac{16.7(S+1.1\pm j3.7)}{(S+.05)(S+1.3)(S+.85\pm j3.6)}$	$\frac{16.7(S+.025)(S+1.0\pm j2.43)(S+1.1\pm j3.74)(S+13)}{(S+.02)(S+.065)(S+1.59)(S+2)(S+3.58)(S+.79\pm j3.78)(S+9.3)}$
Roll	Main Rotor (lateral)	300	$\frac{32.8(S+1.26\pm j4.4)}{(S+.06)(S+1.28)(S+1.2\pm j4.8)}$	$\frac{3.28(S+.025)(S+1.0\pm j2.43)(S+1.25\pm j4.4)(S+13)}{(S+.017)(S+.06)(S+1.56)(S+2)(S+1.6\pm j4.65)(S+6.0\pm j4.23)}$
Yaw	Tail Rotor	1	$\frac{46.4(S-.53)(S+.21\pm j.42)}{S(S-.32)(S+.85)(S-.17\pm j.62)}$	$\frac{46.4(S+.053)(S-.53)(S+.21\pm j.424)(S+1.46\pm j2.3)(S+20)}{S(S+.5)(S+.21\pm j.447)(S+2.5)(S+2.3\pm j2.07)(S+16)}$
Yaw	Tail Rotor	200	$\frac{54.3(S+.14)(S-.05\pm j.49)}{S(S+.05)(S+.85)(S-.17\pm j.62)}$	$\frac{54.3(S-.05\pm j.49)(S+1.4)(S+1.5\pm j2.3)(S+.053)(S+20)}{S(S+.06)(S+1.36)(S+.22\pm j.39)(S+2.5)(S+3.37\pm j4.1)(S+15.3)}$
Yaw	Tail Rotor	300	$\frac{41.6(S+1.58)(S-.15\pm j.78)}{S(S+.06)(S+1.28)(S+1.2\pm j4.8)}$	$\frac{41.6(S+.053)(S-.15\pm j.78)(S+1.57)(S+1.5\pm j2.3)(S+20)}{S(S+.064)(S+1.37)(S+.27\pm j.47)(S+2.5)(S+3.0\pm j5.2)(S+16.6)}$

TABLE X. FIXED-WING TRANSFER FUNCTIONS

Axis	Control	Speed (kts)	Transfer Function, SCAS Off (deg/in.)	Transfer Function, SCAS On (deg/in.)
Pitch	Elevator	200	$\frac{3.74(S+.04)(S+.77)}{(S-.042)(S+.08)(S+.8 \pm j1.1)}$	$\frac{3.74(S+.026)(S+.04)(S+.65 \pm j1.39(S+.77)(S+12.66)}{(S-.026)(S+.53)(S+1)(S+.08 \pm j.02)(S+1.9 \pm j1.12)(S+10)}$
		300	$\frac{8.44(S+.063)(S+1.0)}{(S+.033 \pm j.104)(S+1.15 \pm j2.5)}$	$\frac{8.44(S+.026)(S+.06)(S+.65 \pm j1.39)(S+1.0)(S+12.7)}{(S+.13)(S+.5)(S+1)(S+3.39)(S+.02 \pm j.075)(S+5.64 \pm j4.15)}$
Roll	Ailerons	200	$\frac{121(S+1.01 \pm j3.42)}{(S+.05)(S+1.33)(S+.85 \pm j3.6)}$	$\frac{121(S+1.01 \pm j3.42)}{(S+.019)(S+3.465)(S+.97 \pm j3.62)}$
		300	$\frac{251(S+1.29 \pm j4.57)}{(S+.06)(S+1.27)(S+1.2 \pm j4.77)}$	$\frac{251(S+1.29 \pm j4.57)}{(S+.0126)(S+6.047)(S+1.35 \pm j4.7)}$
Yaw	Rudder	200	$\frac{46.3(S+1.47)(S-.16 \pm j.714)}{S(S+.05)(S+1.33)(S+.85 \pm j3.6)}$	$\frac{46.3(S+1.47)(S-.163 \pm j.714)}{S(S+.143)(S+1.27)(S+1.76 \pm j3.16)}$
		300	$\frac{104.2(S+1.57)(S-.155 \pm j.79)}{S(S+.06)(S+1.27)(S+1.2 \pm j4.77)}$	$\frac{104.2(S+1.57)(S-.155 \pm j.79)}{S(S+.233)(S+1.085)(S+3.3 \pm j3.5)}$

TABLE XI. TRANSFER FUNCTIONS, 200 KNOTS, COMPOUND CONFIGURATION WITH SCAS CONNECTED TO FIXED-WING CONTROLS ONLY (ROTOR CONTROLS OPERATING)

Pitch $$\frac{\theta}{FA} = \frac{11.2\ (S+.04)(S+.77)(S\ .25)(S+.25)(S+12.5)}{(S-.026)(S+.533)(S+.08\pm j.0198)(S+1.91\pm j1.12)(S+10)}$$

Roll $$\frac{\theta}{LAT} = \frac{182\ (S+1.01\pm j3.42)}{(S+.019)(S+3.465)(S+.97\pm j3.62)}$$

Yaw $$\frac{\theta}{Pedal} = \frac{79\ (S+1.47)(S-.163\pm j.714)}{S(S+.143)(S+1.267)(S+1.76\pm j3.16)}$$

TABLE XII. RESPONSE TO ROTOR CONTROL INPUTS (200 KNOTS)

$$\frac{\text{F-A Flapping}}{\text{F-A Cyclic}} = \frac{95900\ (S+11)(S+1.72\pm j14.18)(S+3.38\pm j13.05)}{(S+1.4\pm j14.36)(S+4.19\pm j8.57)(S+6.76\pm j16.1)(S+12.27\pm j37.6)}$$

$$\frac{\text{Lat Flapping}}{\text{F-A Cyclic}} = \frac{53000\ (S-6.6)(S+.99\pm j14.8)(S+3.43\pm j12.8)}{(S+1.40\pm j14.36)(S+4.19\pm j8.57)(S+6.76\pm j16.1)(S+12.27\pm j37.6)}$$

APPENDIX 1

SPECIFICATIONS

DETAIL SPECIFICATION

FOR

ROTOR SYSTEMS RESEARCH AIRCRAFT

1. SCOPE

1.1 This specification establishes the requirement for the following aircraft:

Model designation	Rotor Systems Research Aircraft (RSRA)
Designer's name and model designation	Bell Helicopter Company Model 646
Number of crewmembers	Three
Number and kind of engines	
Main rotor engines	Two Lycoming Model T55-L-7C turboshaft engines
Auxiliary engines	Two Lycoming Model F102-LD-100 turbofan engines
Rotor configuration	
Main	Four-bladed, gimbal-mounted stiff in-plan type
Tail	Four-bladed, gimbal-mounted stiff in-plane type

1.2 Mission. The primary mission of the RSRA is to test new main rotor concepts over their operational envelopes. The test rotors are to be operated to upper stall between the limits of autorotation and maximum obtainable control plane angle of attack for any airspeed between 100 and 200 knots true airspeed (KTAS). With the rotors carried in a minimum-loads position, a high-speed ability of 300 KTAS (sea level and 9500 feet) is desired.

The RSRA is basically a flying wind tunnel, and those moments and forces normally measured in wind tunnels will be measured in flight.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents form a part of this specification to the extent specified herein. In the event of conflict between these documents and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

Military

MIL-W-5013	Wheel and Brake Assemblies, Aircraft
MIL-T-5041	Tire, Pneumatic, Aircraft
MIL-W-5086	Wire, Electric, Hookup and Interconnecting, Polyvinyl Chloride-Insulated, Copper or Copper Alloy Conductor
MIL-B-5087	Bonding, Electrical, and Lightning Protection, for Aerospace Systems
MIL-W-5088	Wiring, Aircraft, Selection and Installation of
MIL-E-5400	Electronic Equipment, Airborne, General Specification for
MIL-H-5440	Hydraulic Systems, Aircraft Types I and II, Design, Installation, and Data Requirements for
MIL-F-5442	Fittings, Push-Pull Engine Control, Quick Disconnect
MIL-C-5503	Cylinders, Aeronautical, Hydraulic Actuating, General Requirements for
MIL-T-5624	Turbine Fuel, Aviation, Grades JP-4 and JP-5
MIL-L-6503	Lighting Equipment Aircraft, General Specification for Installation of

Military (Continued)

MIL-L-6723	Lights, Aircraft, General Specification for
MIL-M-6756	Measuring and Leveling Provisions (for Aircraft)
MIL-E-7016	Electric Load and Power Source Capacity, Aircraft, Analysis of
MIL-E-7080	Electric Equipment, Aircraft Selection and Installation of
MIL-F-7179	Finishes and Coatings, General Specification for Protection of Aerospace Weapons Systems Structures and Parts
MIL-P-7788	Panel, Information, Integrally Illuminated
MIL-L-7808	Lubricating Oil, Aircraft Turbine Engine, Synthetic Base
MIL-C-7958	Controls, Push-Pull, Flexible and Rigid
MIL-H-8501A	Helicopter Flying and Ground Handling Qualities; General Requirements for
MIL-F-8615	Fuel System Components; General Specification for
MIL-S-8698(ASG)	Structural Design Requirements, Helicopters
MIL-I-8700	Installation and Test of Electronic Equipment in Aircraft, General Specification for
MIL-A-8860	Airplane Strength and Rigidity, General Specification for
MIL-A-8870	Airplane Strength and Rigidity Vibration Flutter, and Divergence
MIL-R-8931	Reservoirs, Aircraft and Missile, Hydraulic, Separated Type

Military (Continued)

MIL-F-9490	Flight Control Systems - Design, Installation and Test of, Piloted Aircraft, Central Specification for
MIL-W-16878	Wire, Electrical, Insulated, High Temperature
MIL-C-18244	Control and Stabilization Systems, Automatic, Piloted Aircraft, General Specification for
MIL-F-18372	Flight Control Systems
MIL-P-19692	Pumps, Hydraulic, Variable Delivery, General Specification for
MIL-L-23699	Lubricating Oil, Aircraft Turbine Engines, Synthetic Base
MIL-C-25050	Color, Aeronautical Lights and Lighting Equipment, General Requirements for
MIL-L-25467	Lighting, Integral, Aircraft Instrument, General Specification for
MIL-E-25499	Electrical Systems, Aircraft, Design and Installation of General Specification for
MIL-S-38039	System, Illuminated, Warning, Caution and Advisory, General Specification for
MIL-F-38363	Fuel System, Aircraft, Design Performance, Installation, Testing, and Data Requirements, General Specification for
MIL-W-81044	Wire, Electric, Crosslinked Polyalkene Insulated, Copper
MIL-T-81259	Tie-down, Airframe, Design, Requirements for

Military (Continued)

MIL-H-83282	Hydraulic Fluid, Fire Resistant Synthetic Hydrocarbon Base, Aircraft

STANDARDS

Military

MIL-STD-130	Identification Marking of U. S. Military Property
MIL-STD-143	Standards and Specifications, Order of Precedence for the Selection of
MIL-STD-250	Aircrew Station Controls and Displays for Rotary Wing Aircraft
MIL-STD-411	Aircrew Station Signals
MIL-STD-704	Electric Power, Aircraft, Characteristics and Utilization of
MIL-STD-805	Towing Fittings and Provisions for Fixed-wing Aircraft
MIL-STD-809	Adapter, Aircraft, Jacking Point, Design and Installation of
MIL-STD-838	Lubrication of Military Equipment
MIL-STD-850	Aircrew Station Vision Requirements for Military Aircraft
MIL-STD-1333	Aircrew Station Geometry for Military Aircraft

DRAWINGS

Military Standards

MS33785	Instrument Arrangement, Standard Basic for Fixed and Rotary Wing Aircraft

PUBLICATIONS

Military

MIL-HDBK-5	Metallic Materials and Elements for Aerospace Vehicle Structures

Department of Army

ADS-3	Aeronautical Design Standard, U. S. Army Anthropometric Data
TB 746-93-2	Painting and Marking of Army Aircraft
TM 55-6600-200-20	Marking of Instruments and Interpretation of Markings

2.2 Non-Government documents. The following documents form a part of this specification to the extent specified herein. In the event of conflict between these documents and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

Lycoming

124.40	Engine Specification, F102-LD-100, Turbofan Engine
124.43	Engine Specification, T55-L-7C Turboshaft Engine

DRAWINGS

Bell Helicopter Company

646-900-001	Inboard Profile RSRA

REPORTS

Bell Helicopter Company

646-099-003	RSRA Development Plan

3. REQUIREMENTS

3.1 Aircraft characteristics.

3.1.1 Drawings.

3.1.1.1 Three-view/inboard profile drawing. Refer to BHC Drawing 646-900-001 in Appendix 2.

3.1.1.2 Main subassembly drawings. Refer to Appendix 2.

3.1.2 Aircraft performance.

3.1.2.1 Required performance (compound configuration).

a. The aircraft shall be capable of hovering OGE for 30 minutes at not more than military rated power at sea level (standard day, or 95°F), flying for 10 nautical miles, performing 2 minutes of hover, and landing with required fuel reserves (see 3.1.2.2.1).

b. The aircraft shall be capable of flying at 300 KTAS for 15 minutes at either sea level, standard day, or at 9500 feet, standard day, while carrying a removable payload of 2000 pounds, and then flying for a distance equal to that required to accelerate to 300 KTAS, hover for 2 minutes OGE, and landing with the required fuel reserve.

c. The installed auxiliary propulsion system shall be capable of driving the aircraft to 300 KTAS at sea level with the drag produced by an autorotating specific design rotor (whether or not the rotor is capable of autorotation at 300 KTAS).

d. The wing shall be capable of supporting the design gross weight of the aircraft at 150 KEAS at sea level, with flaps retracted.

3.1.2.2 Expected performance (compound configuration).

a. The aircraft shall be able to hover OGE for 30 minutes at sea level, standard day, while carrying a removable payload of 3200 pounds (2400 pounds at 95°F, sea level) and not exceed the transmission limits. Expected hover performance is shown in Figure 5 of the basic report.

3.1.2.2 (Continued)

b. The aircraft shall be able to maintain 300 KTAS for 30 minutes at sea level, standard day, while carrying 1000 pounds of removable payload (22 minutes carrying 2000 pounds) and complying with the requirements of the high-speed mission. Expected sea level, standard day performance at 300 KTAS is shown in Figure 3 of the basic report.

c. The aircraft shall be able to maintain 300 KTAS for 30 minutes at 9500 feet, standard day, while carrying 1000 pounds of removable payload (20 minutes carrying 2000 pounds) and complying with the requirements of the high-speed mission. Expected performance at 9500 feet, standard day is shown in Figure 4 of the basic report.

3.1.2.2.1 Fuel allowance. All required missions shall allow fuel for 2 minutes of operation of all engines at normal rated power for warmup and takeoff, and 2 minutes of flight at minimum attainable airspeed immediately after take-off. Fuel reserve on landing shall be that which will allow 20 minutes cruise at the speed for best range, or 10 percent of takeoff fuel, whichever is greater.

3.1.3 Weights.

3.1.3.1 Group weight statement. An estimated weight breakdown by major components and systems is provided in Table II of the basic report.

3.1.3.2 Mission gross weights. Gross weights for both the primary (15-minute) and the alternate (30-minute) high speed mission are provided in Table III of the basic report.

3.1.4 Center-of-gravity data. Center of gravity locations for the compound and pure helicopter configuration are shown in Table IV of the basic report.

3.1.5 Areas. The principal areas are as follows (this information is not to be used for inspection purposes):

Main rotor blade (total)	229.2 sq ft
Main rotor geometric disc (total)	2380.0 sq ft
Main rotor blade geometric solidity	0.094
Wing (total, including fuselage carry-through)	225.0 sq ft
Flap (total, upper and lower inboard panels and outer lower panels)	128.9 sq ft

3.1.5 (Continued)

Roll control surface	10.9 sq ft
Stabilator	50.0 sq ft
Vertical stabilizer	21.2 sq ft
Rudder	4.3 sq ft
Ventral fin	17.5 sq ft
Tail rotor blade (total)	17.5 sq ft
Tail rotor geometric disc (total)	78.54 sq ft
Tail rotor geometric solidity ratio	0.191

3.1.6 Dimensions and general data. The principal dimensions and general data are as follows (this information is not to be used for inspection purposes):

MAIN ROTOR

Number of blades	4
Diameter	55 ft 0 in.
Blade chord	2 ft 1.0 in.
Blade twist angle	0 deg
Disc loading at design gross weight of 23,740 lb	9.98 psf
Airfoil section designation	Wortmann FX090 (Modified)
Thickness	
At root	12.7 percent
At tip	9.0 percent
Engine-to-main-rotor drive ratio	23.1:1

3.1.6 (Continued)

WING

Span	30 ft 0 in.
Chord	
At fuselage center line	10 ft 0 in.
At tip	5 ft 0 in.
Airfoil section designation and thickness	65_3A618
Incidence range	±20 deg
Quarter chord sweep angle	2.85 deg
Aspect ratio	4.0
Taper ratio	2.0
Flaps, upper and lower inboard panels and lower outboard panels	
Percent of chord	30 percent
Percent of span	
Lower	100 percent
Upper	60 percent
Maximum deflections	
Inboard surfaces - as drag producers	60 deg
Outboard lower surface	30 deg
Roll control surfaces	
Percent of span	40 percent
Maximum deflection	30 deg

3.1.6 (Continued)

EMPENNAGE

Stabilator

Span	14 ft 1.2 in.
Mean geometric chord (at B.L. 38.0)	3 ft 7.8 in.
Airfoil section designation and thickness	NACA 0012
Maximum deflection	±15 deg

Vertical fin

Span	6 ft 3.0 in.
Mean geometric chord (at B.L.O)	3 ft 11.0 in.
Airfoil section designation and thickness	NACA 0012
Sweep angle of quarter chord	42 deg
Aspect ratio	4.0
Taper ratio	2.0

Rudder

Percent chord	30 percent
Percent span	50 percent
Maximum deflection	± 30 deg

Ventral fin

Span	3 ft 6.0 in.
Mean geometric chord	4 ft 9.0 in.
Airfoil section designation and thickness	NACA 0012
Sweep of quarter chord	38 deg
Aspect ratio	0.75

3.1.6 (Continued)

TAIL ROTOR	
Number of blades	4
Diameter	10 ft 0.0 in.
Blade chord (constant)	10.5 in.
Blade twist angle	0 deg
Airfoil section designation	Wortmann FX083 (Modified)
Thickness	
At root	23.4 percent
At tip	8.3 percent
Engine-to-tail-rotor drive ratio	4.13:1
LANDING GEAR	
Main	
Tread	9 ft 4.0 in.
Tire size	24 x 8
Oleo strut travel	12.0 in.
Tail wheel	
Tire size	18 x 4.4
Oleo strut travel	4.7 in.
Axle travel	12.0
WIDTH	
Main rotor blades turning	55 ft 0 in.

3.1.6 (Continued)

LENGTH

Maximum-main rotor blades at rest, one trailing, main rotor fwd tip to tail rotor aft tip, 4° mast tilt	66 ft. 5 in.
Maximum-most forward part of nose aft part of vertical fin	51 ft. 10 in.
From centerline of main rotor to horizontal tail MAC quarterchord point at aircraft centerline	20 ft. 6 in.

HEIGHT (three-point attitude)

Over main rotor mast (4° mast tilt)	15 ft. 0 in.
Over highest part of tail	10 ft. 0 in.
Over highest part of fuselage sail	12 ft. 2 in.
In level attitude, from top of mast to bottom of wheels (no load on wheels)	15 ft. 3 in.
Minimum main rotor clearance (ground to tip, rotor turning, 3-point attitude zero flapping, mast tilt 12° fwd)	10 ft. 11 in.
Minimum main rotor clearance (tail rotor to tip, rotor turning 12° flapping, mast tilt 4° fwd)	1 ft. 6 in.
Tail rotor clearance (ground to tip, rotor turning)	3 ft. 8 in.

3.1.7 <u>Control surface and corresponding control movements.</u> Control surfaces and corresponding control movements as limited by control stops are as follows (this information is not to be used for inspection purposes):

Range of main rotor blade root collective pitch angles	0 to 18 deg
Collective pitch control lever travel	11.25 in. up 0 in. down

3.1.7 (Continued)

Maximum range of main rotor blade with angles with cyclic pitch (collective pitch at low position)	14 deg fwd 12 deg aft 10 deg right 10 deg left
Cyclic pitch stick travel Longitudinal	 5.50 in. fwd
Lateral	5.00 in. right 5.00 in. left
Flap, (inboard) maximum deflection	60 deg
Flap, (outboard), maximum deflection	30 deg
Ailerons (upper), maximum deflection	30 deg
Aileron control travel	± 5.00 in.
Stabilator, maximum deflection	± 15 deg
Stabilator control travel	± 5.50 in.
Rudder, maximum deflection	± 30 deg
Rudder pedal travel	3.25 in. fwd 3.25 in. aft
Range of tail rotor blade angles (root)	-13 to +24 deg

3.2 General features of design and construction

3.2.1 General interior arrangement. The interior arrangement of the aircraft shall provide seating for three: a pilot, copilot/computer operator, and a third crew member. The cabin arrangement shall accommodate the 5th through 95th percentile Army Aviator and defined in ADS-3. The copilot will be seated forward of the pilot's station on the aircraft's centerline. The pilot and the third crew member will be seated in a side-by-side configuration with the pilot seated on the right.

3.2.2 Selection of materials and parts. Materials, processes, and parts used in the manufacture of the research aircraft shall be of high quality, suitable for the purpose, and shall conform to applicable Government specifications where possible. Specifications and standards for all materials, parts, and processes which are not specifically designated herein and which are necessary for the execution of this specification shall be

3.2.2 (Continued)

selected in accordance with MIL-STD-143. Military standard parts shall be used whenever they are suitable for the purpose. Any standard part, of the same part number, shall be readily interchangeable as applied within the aircraft. Commercial utility parts such as screws, bolts, nuts, and cotter pins may be used provided they possess suitable properties.

3.2.3 Workmanship. Workmanship shall be in accordance with high-grade aircraft practice and of quality to ensure safety, proper operation, and service life.

3.2.4 Production, maintenance and repair. The design of the aircraft, insofar as a research aircraft is concerned, shall be such as will ensure ease of manufacture, rapid installation of power plant and transmissions, and ease of general maintenance. Special attention shall be given to the case with which the component parts of the structure and installation can be inspected, maintained, and repaired. A minimum of special tools will be required for component changes. All required ground power equipment shall be military standard or commercially available.

3.2.5 Interchangeability and replaceability. Interchangeability shall not be required, however, replaceability of major parts and assemblies shall be maintained insofar as possible on a research aircraft.

3.2.6 Finish. Finish and protective coatings shall be as specified in the applicable BHC Finish Specification prepared in accordance with MIL-F-7179.

3.2.6.1 Exterior color. The exterior color shall be in accordance with the research and development aircraft requirements of TB 746-93-2.

3.2.6.2 Interior color. The interior color shall be in accordance with the research and development aircraft requirements of TB 746-93-2 as specified in the applicable BHC Finish Specification.

3.2.7 Identification and marking. The aircraft and its components shall be identified and otherwise marked in accordance with the applicable provisions of MIL-STD-130 and TB 746-93-2 for research aircraft.

3.2.8 Extreme temperature operations. The aircraft shall be designed for operation within the ambient temperature range of -25° to +125°F.

3.2.9 Climatic requirements. The aircraft and its equipment shall not be adversely affected by other climatic conditions incident to the temperature range outlined in 3.2.8 and shall be capable of transfer from one climate to another without penalty of extensive modification and adjustment.

3.2.10 Lubrication. Aircraft lubricants shall conform to the requirements of MIL-STD-838.

3.2.11 Equipment and furnishings installation. The equipment and furnishings specified herein shall be contractor furnished except the flight control computer which shall be Government furnished.

3.2.12 Noise and vibration requirements. The aircraft and its equipment shall function normally in all extremes of noise and vibration that will be encountered.

3.2.13 Reliability. The aircraft and its related subsystem shall be designed with maximum reliability as a goal.

3.2.14 Maintainability. The aircraft and its related subsystems shall be designed with maximum maintainability as a goal. Special consideration will be given to provisions for access, rapid removal, and replacement of components.

3.3 Aerodynamics.

3.3.1 Aerodynamics design. The configuration of the aircraft shall be such as to make efficient use of the current state-of-the-art in aerodynamic design. Reasonable compromises shall be permitted to make the aircraft and installed equipment satisfy the design requirements or characteristics.

3.3.2 Stability and control.

3.3.2.1 Control characteristics. The requirements of MIL-H-8501A shall be used as a design guide for the stability and control characteristics of the aircraft. The loss of a thrust engine in flight will result in a reduced maximum speed ability, but will not create a pitch trim problem. The loss of a main rotor engine in hover or in low speed flight will result in safe height-velocity restrictions similar to current twin-engine helicopters.

3.4 Structural design criteria.

3.4.1 Strength requirements. Strength and rigidity shall be provided using MIL-A-8860 and MIL-S-8698(ASG) as design guides. The allowable stress values of MIL-HDBK-5 shall be used as a guide. Conventional helicopter and subsonic aircraft materials and structure will be used.

3.4.2 Design limit load factor. The design limit load factor at the design gross weight shall be as follows:

Positive (+) 4 Negative (-) 1.5

To obtain an ultimate load factor the limit load factor is multiplied by 1.5 (a factor of safety).

3.4.3 Design speed. The aircraft structure shall be designed for a speed of 360 KTAS at sea level conditions.

3.4.4 Sink speed. The aircraft shall be designed for a limit sinking speed of 10 feet per second (fps) from a hover in combination with two-third's of the hovering thrust at the design gross weight. The limit sinking speed for conventional landing shall be 10 fps with lift equal to weight at the design gross weight.

3.4.5 Structural design service life. The design of all primary structure shall be on a fail-safe basis to the maximum extent feasible for a research aircraft program. The service life of the aircraft shall be a minimum of 600 hours.

3.4.6 Flutter characteristics. The aircraft shall be free from divergence, flutter, buzz, or other instability throughout its range of design speeds, altitudes, maneuvers, and loading and weight conditions. Flutter requirements shall be determined using MIL-A-8870 as a guide.

3.5 Main rotor group. The main rotor shall be a four-bladed, gimbal-mounted, stiff-in-plane type with a hub moment spring. The main rotor group shall consist of the yoke, pitch change bearings, blade restraint bearings, grips, blades, gimbal assembly, and hub spring. The main rotor shall attach to the mast by means of the drive plate of the gimbal assembly.

3.5.1 Blade construction. The blade airfoil shall be of the FX (Wortmann) series. The main rotor blade shall consist of a leading edge spar, afterbody skins, honeycomb core, trailing edge strip, root-end doublers and grip plates bonded together. All structural materials shall be corrosion-resistant steel except the honeycomb core which shall be aluminum alloy with a corrosion-resistant coating. A single swept tip shall be provided for improved aerodynamic efficiency. The blades shall be individually balanced and individually interchangeable.

3.5.2 Blade retention. The blades shall be attached to the grip by means of a two-bolt attachment.

3.5.3 Rotor head. The main rotor head shall consist of the hub assembly which is comprised of the yoke, blade grips, pitch change bearings, blade retention bearings, gimbal assembly, and hug spring. Bearings shall not require lubrication.

3.5.4 Rotor head controls. The main rotor controls shall consist of those components required to transfer pilot or computer-initiated input motions to the rotor head as collective or cyclic pitch.

3.5.4.1 Swashplate assembly. The swashplate assembly shall consist of a rotating outer ring mounted through a pair of ball thrust bearings to a nonrotating ring mounted on a ball type support. Tilting the assembled rings on the ball transmits cyclic pitch motion through connecting pitch links to the main rotor head pitch horns. Raising or lowering the swashplate vertically delivers collective pitch to the rotor.

3.5.4.2 Cyclic and collective mixing system. The cyclic and collective mixing system shall consist of nonrotating levers assembled below and connected to the swashplate to provide control mode mixing to the swashplate.

3.5.4.3 Swashplate drive system. The swashplate drive system shall consist of a mast mounted hub connected through links to the swashplate rotating ring to provide rotary drive motion.

3.5.5 Flapping stops. A flapping stop shall be provided in the gimbal to limit rotor flapping to 10 degrees. The yoke shall incorporate a flex plate which provides coning relief.

3.5.6 Blade tracking and balancing. Blades shall be individually balanced to accurate standards to permit initial run-up after installation without complete rotor balancing. Blades shall be manufactured to close geometric tolerances so that tracking and balancing operations on the aircraft are minimized.

3.6 Wing group. The aircraft shall have a variable incidence, cantilevered wing with movable trailing-edge surfaces. This removable wing shall be attached to the fuselage carry-through torque box by means of bolted flanges. The upper and lower trailing-edge surfaces of each wing panel are split and shall serve as lift, drag, and roll control devices.

3.6.1 Construction. The wing airfoil section shall be a 65_3A618 with a 30-foot span, a 10-foot centerline chord, and

3.6.1 (Continued)

a 5-foot tip chord. Each wing panel shall be constructed of two main spars with aluminum spar caps and aluminum-faced honeycomb webs and skins. Integral fuel cells shall be provided in each wing panel which will provide a total wing capacity of 1700 pounds of fuel. The fuel cells shall be of crashworthy construction.

3.6.2 Variable wing incidence. Variable wing incidence shall be provided by tilting the wing about a trunnion mounted to the wing balance. The wing incidence shall be controllable in flight by the pilot by the means of an electromechanical screw jack connected to the wing pitch control arm. The incidence range shall be ±20 degrees.

3.6.3 Lift and drag devices. The outboard lower trailing edge surface shall serve as flaps. The inboard upper and lower trailing edge surfaces shall serve as flaps and high-drag devices. The inboard panels can be operated individually as flaps and collectively as drag devices. The upper inboard panel shall be driven by a single hydraulic motor. The lower inboard panels are driven by either of two hydraulic motors, with the other motor serving as a standby. The travel limits for the lower outer panel is 30 degrees and for the inboard upper and lower panel is 60 degrees. A position indicator shall be provided for the lift and drag devices.

3.6.4 Roll control devices. The outboard upper trailing edge surface shall serve only as a roll control device. This panel shall be driven by a dual hydraulic actuator and controlled by pilot motion of the cyclic stick. Travel limit for the panel is 30 degrees.

3.7 Antitorque (tail) rotor group. The tail rotor shall be a four-bladed, gimbal-mounted, stiff-in-plane type. The tail rotor group shall consist of the gimbal, yoke, pitch change bearings, blade restraint bearings, and blades. The tail rotor shall be mounted on the aircraft such as to be protected from contact with objects during landings, and to minimize hazard to ground personnel. The tail rotor and tail rotor drive train shall be capable of producing 120 percent of the thrust required to counteract main rotor torque when the main rotor is operating at transmission torque limit. Rearward or sideward flight is not required except the aircraft shall be capable of hovering in winds up to 12 knots from any direction.

3.7.1 Blade construction. The blade airfoil shall be of the FX(Wortmann) series. The tail rotor blade shall consist of a leading edge spar, afterbody skins, honeycomb core,

3.7.1 (Continued)

trailing edge strip, root-end doublers and grip plates bonded together. The blade spar shall be of stainless steel sheet, stretch-formed to contour. The blade skins shall be aluminum alloy sheet. Use of the stainless steel leading edge will provide protection against erosion environments in the area where abrasion removes protective coatings. The blade shall be tapered in thickness; the thick root shall house the pitch change bearings which attach the blade to the yoke. Aluminum honeycomb core shall be used to stabilize the afterbody skins. A swept tip for improved aerodynamic efficiency and noise reduction shall comprise the outboard end of the blade. This tip shall consist of an aluminum die casting protected with a stainless steel erosion shoe and a stainless steel aft tip fairing. The blades shall be individually balanced and individually balanced and individually interchangeable.

3.7.2 Rotor head. The rotor head shall consist of the gimbal, yoke, pitch change bearings, and blade restraint bearings. All bearings shall be of the non-lubrication type.

3.7.3 Rotor head control. Controllable collective pitch is provided to the tail rotor. The control system shall consist of a crosshead driven by the mast. Control motion shall be transmitted to the crosshead through a grease lubricated angular contact bearing set. An A-frame bellcrank shall connect the outer bearing housing to the fixed control system. Pitch control rods shall be employed to connect the crosshead to the rotor blade pitch horn.

3.7.4 Blade tracking and balancing. The tail rotor shall be statically balanced as an assembly; final adjustment shall be made on the aircraft with the rotor operating at normal speed. Tracking shall be accomplished under the same conditions.

3.8 Tail group. The tail group shall consist of stabilator, vertical fin, rudder, and ventral fin.

3.8.1 Stabilator. The controllable stabilator shall be attached to the tailboom by means of antifriction bearings and bolts. The stabilator shall be structurally adequate to withstand aerodynamic loads. It shall be synchronized with the fore and aft cyclic control, but may be independently controlled by a limited authority SCAS.

3.8.2 Vertical fin and rudder.

3.8.2.1 Vertical fin. The swept vertical fin shall be fixed to the aft end of the tailboom and extend upward. The fin shall include an aerodynamic fairing around the tail rotor gearbox.

3.8.2.1 (Continued)

It shall be a two spar structure with aluminum alloy ribs and skin. Hinge fittings shall be carried on the rear spar for the rudder. A tail rotor balance mounting platform will be mounted forward of the front spar.

3.8.2.2 Rudder. The rudder shall attach to the vertical fin through hinge fittings. Control fittings shall also be attached to the rudder to provide synchronized control with the rudder pedals.

3.8.3 Ventral fin. The ventral fin shall be fixed to the aft end of the tailboom and extend downward. The lower portion of the fin will house the tail wheel for steering. The fin shall be constructed of aluminum alloy skin and stiffners.

3.9 Body group. The body group shall consist of two main landing gears, wing attaching area, engine section, rotor pylon assemblies, and equipment compartments.

3.9.1.1 Construction. The fuselage shall be a conventional nonpressurized semimonocoque structure. Aluminum alloy construction shall be extensively employed in the primary structure of the fuselage. Aluminum honeycomb may also be employed.

3.9.1.2 Crew compartment. The crew compartment shall accommodate the 5th through 95th percentile Army aviator as defined in ADS-3. The geometry of the flight crew stations shall be in accordance with MIL-STD-1333. Crew compartment interior colors shall be in accordance with TB 746-93-2.

3.9.1.2.1 Crew compartment enclosures. The crew compartment enclosures shall extend from the nose of the aircraft to the front of the rotor pylon compartment. The enclosures shall be water tight. The combination of transparent canopy materials and support frames shall be sufficiently strong to withstand loads to 360 KTAS at sea level. The canopy shall be mounted to permit thermal expansion/contraction without distortion of any transparent panel or its frame.

3.9.1.2.2 Field of view. Maximum practicable vision shall be provided for the crew. Radii-of-curvature and angles-of-incidence shall be employed consistent with aerodynamic, structural, and fabricating considerations, which will result in the least possible optical distortion in the transparent components, and prevent reflections of objects both within

3.9.1.2.2 (Continued)

and without the cockpit from interfering with the pilot's vision. The provisions of MIL-STD-850 shall be used as a design guide.

3.9.1.2.3 Entrance. The canopy shall be provided with movable sections which will readily permit normal entrance and exit of the crew and necessary equipment. The doors shall be hinged at the top and shall have supports for incorporating locks to hold the doors open. The doors shall be provided with a seal to prevent entrance of sand, dust, or spray. Joints of doors shall be smooth with no gaps to cause a breakdown of airflow. Latches, hinges, and locks shall be of corrosion-resistant material or adequately protected against corrosion.

3.9.1.2.4 Emergency escape system. A YANKEE emergency escape (extraction) system shall be provided for the crew. The extraction method shall be compatible for crew members with full personnel flight gear. Component parts of the system shall be the rotor and canopy severance devices, canopy thrusters, sequencer, extraction rockets and their launchers, and a terrain looker. The rotor severance device ballistically separates the rotor system from the aircraft. The canopy severance device ballistically separates the canopy into two pieces and the thrusters separate the two canopy pieces from the aircraft. The terrain looker prevents the extraction of the crew in the direction of the earth or another helicopter, and serves as a safety device to prevent inadvertent extraction through the rotor disc. A single emergency escape actuation control will, when activated, initiate an automatic escape sequence which will insure all required escape functions are performed. In addition to automatic sequencing, independent removal of the rotor, canopy, or both shall be provided. The system actuation controls shall be designed and located to minimize the possibility of their inadvertent actuation.

3.9.1.3 Equipment compartments. Two equipment compartments shall be provided for the accommodation of a GFE computer, data recording, instrumentation and telemetry equipment, and avionic and electrical equipment. External access doors shall be provided for routine service, checkout, and maintenance of the equipment. One compartment shall be a minimum of 15 cubic feet for avionic and electrical equipment and one a minimum of 20 cubic feet for computer and data acquisition equipment.

3.9.2 Tailboom. The tailboom shall be a conventional semi-monocoque plate-stringer structure, using frames, stringers, and longerons. The tailboom shall attach to the fuselage section by bolts. The tail rotor driveshaft shall be located in the upper portion of the tailboom and shall be housed within an easily opened fairing.

3.10 Landing gear. The landing gear shall be a conventional type with two fully retractable main gears and a nonretractable tail wheel. The landing gear shall be capable of ground taxi, towing, ground handling, take off and landing, including landings at the design sink speed of 10 fps, and at forward speeds from hover to 150 KTAS.

3.10.1 Main gear. The main landing gear shall consist of right- and left-hand units mounted on the mid-fuselage section and shall retract forward into the fuselage. Mechanical down and up locks with crew compartment indicators shall be provided. An emergency pneumatic gear extension system shall be provided to permit gear extension in the event of utility hydraulic system failure.

3.10.2 Tail wheel. A nonretractable tail wheel shall be provided at the aft end of the tailboom. The tail wheel shall be free to caster when unlocked. The tail wheel shall absorb energy from tail low landings and protect the tail rotor from ground impact. The upper portion of the tail wheel shall be housed by the lower portion of the ventral fin, which pivots with the tail wheel.

3.10.3 Shock absorbers. The landing gear shock absorbing device shall be a single stage air/oil oleo strut. For oleo strut travel reference 3.1.6.

3.10.4 Braking system. Brakes shall be in accordance with MIL-W-5013. The system shall have an energy absorption capacity adequate for development of 8 fps^2 deceleration (without rotor braking) for a landing at design gross weight with flight idle thrust on auxiliary engines, at a touch-down airspeed of 150 KTAS. Toe-operated brake pedals shall be provided for the pilot and copilot. Parking brakes shall also be provided. The system shall be powered by the utility hydraulic system. In the event of utility system failure, pneumatic emergency brake actuation shall be provided for a limited number of brake applications.

3.10.5 Wheels and casings. Main wheels shall be in accordance with MIL-W-5013. Pneumatic tubeless casings shall be in accordance with MIL-T-5041. For tire size reference 3.1.6.

3.11 Flight control group. Design and construction of the flight control group shall be in accordance with MIL-F-9490, MIL-C-18244, MIL-F-18372 and MIL-STD-250. Design shall be such that all control rods, hydraulic lines, and other critical items are positioned or supported to prevent exposure or damage by personnel during servicing and maintenance. It shall be possible for the pilot to maintain satisfactory control in autorotational landings in the event of complete engine failure, or in flight as a pure helicopter, fixed-wing aircraft, or autogyro.

3.11.1 Flight control system. The flight control system shall be designed to accommodate the 5th and 95th percentile man. It shall consist of primary controls, mechanical linkages to rotors and fixed-wing control surfaces, and parallel electrical and electronic backup system and augmentation equipment. The arrangement, location, and actuation of controls and related items of equipment shall be in accordance with MIL-STD-250.

3.11.2 Helicopter flight control subsystem. The helicopter flight control subsystem shall provide lateral, longitudinal, directional, and vertical controls in accordance with MIL-F-18372. Positive stops shall be provided in the control system to prevent movement of the rotor controls beyond safe limits.

3.11.2.1 Cyclic controls (lateral and longitudinal). Three cyclic control sticks shall be provided, one forward of each crew position. Each cyclic control stick shall provide both lateral and longitudinal control of the main rotor. Both the lateral and longitudinal systems shall consist of hydraulic servo-units, an electro-hydraulic force-feel device, push-pull tubes, a mixing device, and bellcranks. All cyclic sticks shall be provided with cyclic force-feel trim. Force-feel release, radio/ICS, SCAS release and cyclic washout control switches will be incorporated on the pilot and copilot sticks.

3.11.2.2 Collective pitch controls (vertical). Two collective control levers shall be provided, one at the left of the pilot, and one at the left of the copilot. The collective pitch control shall incorporate a hydraulic servo-unit, an electro-hydraulic stick force-feel device, collective and tail rotor washout control switches, a friction device, push-pull tubes, and bellcranks.

3.11.2.3 Tail rotor pitch control (directional). Three sets of directional control pedals shall be provided, one set forward of each crew position. The system shall consist of adjustable pedals, bellcranks, and push-pull tubes, and a hydraulic servo-unit. The pilot's and copilot's pedals shall be provided with toe brakes.

3.11.2.4 Wash-out controls. A means shall be provided to portionally wash-out the main- and tail-rotor controls, so that the collective and cyclic control sticks and directional control pedal motions will not produce movement of the rotor controls during high-speed flight. Three-position, spring-loaded "off" switches are provided on the control sticks for wash-out control. Visual presentation of the degree of wash-out shall be provided. The wash-out device shall be mechanically operable in the event of electrical failure.

3.11.3 Fixed-wing flight control system. The fixed-wing flight control system shall be integrated with the conventional helicopter flight controls. Wing roll control devices, the stabilator, and the rudders shall be commanded by the same pilot controls that provide lateral cyclic, longitudinal cyclic, and tail rotor collective control respectively.

3.11.3.1 Roll control. Rolling moments shall be provided by a split trailing-edge control located on the outer 40 percent of the wing's upper surface. Control of the surface shall be provided by a dual servo-unit, commanded by lateral cyclic stick motions and tandem SCAS actuators.

3.11.3.2 Pitch control. Pitching moments shall be generated by the stabilator to provide longitudinal control. Control of the stabilator shall be provided by longitudinal cyclic stick motions and tandem SCAS actuators.

3.11.3.3 Sideslip control. Yawing moments shall be generated by the rudder to provide sideslip control. Control of the rudder shall be provided by directional control pedal commands and tandem SCAS actuators.

3.11.3.4 Drag device control. The wing's upper- and lower-inboard split flaps will open together to produce drag. A three-position, spring loaded "off" switch shall be mounted on the auxiliary thrust engines throttle quadrant for drag control. Visual presentations of the position of all wing trailing-edge surfaces (except roll control) shall be provided on the instrument panels. Deployment of drag devices will not result in any abrupt or uncontrollable trim changes.

3.11.3.5 High lift device control. The wing's lower surface flaps (both inboard and outboard) shall lower together to produce high lift. The wing's upper surface inboard flaps shall open to provide rotor download. Instrument panel switches shall be provided for control of the flaps.

3.11.4 Electronic flight control system. An electronic flight control system shall be provided which will be compatible with the GFE flight control computer. The electronic system shall exercise control of the rotor controls, fixed-wing controls, and auxiliary thrust engines. Design of the system shall be such that failures will result in slow centering of the affected series actuators. Other actuators will be bypassed. The electronic system will operate all flight controls through the SCAS actuators, and will provide force-feel for the cyclic stick, when control operation is in the manual mode. All SCAS functions will be accomplished by the electronic flight control system.

3.11.5 Hydraulic power. The flight controls shall be powered by a dual hydraulic system (see 3.14.2). Operation of the manual flight controls shall be possible with complete hydraulic failure.

3.11.6 Electrical power. The manual flight control system shall be completely operable in event of total electrical failure.

3.12 Engine section.

3.12.1 Main engine section. The main engine section shall consist of the upper part of the aft portion of the fuselage. This section shall include engine mounting provisions, firewalls, bulkheads, and cowlings. It shall be constructed of a deck, supported by beams, bulkheads, and skin. The engines shall be mounted side-by-side in a horizontal position. The mounts shall consist of built-up steel members and fittings. Access doors and removable cowling shall be provided normal inspection and maintenance. Provisions will be included which will allow engine movement or installation of other types of engines.

3.12.2 Auxiliary engines section. The auxiliary thrust engines shall be mounted on fuselage-attached pylons and housed within individual cowlings. Thrust engines and pylons will be easily removable, leaving a smooth fuselage area. Access doors and removable cowling shall be provided for normal inspection and maintenance.

3.12.3 Fire protection. Both the main engine and the thrust engines shall be provided with fire detection and extinguishing systems controllable from the pilot's position.

3.13 Propulsion subsystem.

3.13.1 Description. The propulsion subsystem shall include the main propulsion engines, auxiliary thrust engines, main engine driven accessories, air induction system, exhaust system, cooling system, lubricating system, fuel system, propulsion system controls, starting system, and main rotor drive system.

3.13.2 Main propulsion engines. The main propulsion engines shall be two Lycoming T55-L-7C shaft turbine engines conforming to Lycoming Specification 124.43, with a maximum power rating of 2650 shaft horsepower per engine. The engines shall be horizontally-mounted side-by-side and shall drive the transmission through an interconnecting combining gearbox.

3.13.3 Auxiliary thrust engines. The auxiliary thrust engines shall be two Lycoming F102-LD-100 conforming to Lycoming Specification 124.40 with a maximum static thrust of 7200 pounds per engine. The thrust engines shall be mounted on pylons, one attached to each side of the fuselage.

3.13.4 Engine driven accessories. The following aircraft components are driven by the main engines or auxiliary thrust engines, as applicable:

a. Main engines

(1) Tachometer generator (gas producer)

(2) Tachometer generator (power turbine)

(3) Combining gearbox

(4) Starter generator

b. Auxiliary thrust engines

(1) Starter

(2) Tachometer generator

3.13.5 Air induction system. The air induction system shall consist of an aerodynamically-smooth opening in the cowl to allow entrance of air for the engines. The system shall have sufficient strength so that vibration and flexing normally expected in service will not cause damage. The design of the induction inlet and duct shall not cause any erratic or adverse air flow distribution which would cause engine compressor stall or other engine malfunction at any condition and attitudes within the aircraft design operation envelope. Interior duct surfaces shall be aerodynamically smooth. The installation shall be such that foreign objects cannot be retained in the joints.

3.13.6 Exhaust system. The exhaust system shall include the tailpipes, together with the necessary flanges, supports, and attachments required to carry off exhaust gases for each engine. The exhaust system shall be constructed of materials resistant to heat and corrosion. Exhaust gases shall be ducted upward and away from the aircraft structure. Two-position exhaust nozzles, controllable from the pilot's position, will be fitted on both main rotor engines to reduce drag at high forward speeds.

3.13.7 Cooling system. The engine cooling system shall prevent the temperatures of the engine, engine-mounted accessories, engine components, equipment located in the engine compartment, associated structure, and engine fluids from exceeding the allowable limits.

3.13.8 Lubricating system. The lubrication system shall be self-contained within the engines. A reservoir, oil cooling provisions, lubrication pumps, and oil filters shall be provided. Oil pressure and oil temperature gages shall be provided for the engines. The engines shall function satisfactorily throughout their operating envelope when using oil conforming to and having any of the variations in characteristics permitted by MIL-L-7808 or MIL-L-23699.

3.13.9 Fuel system. The fuel system shall consist of six interconnected crashworthy fuel cells, four in the fuselage and one in each wing. The system shall include fuel pumps, shutoff valves, fuel quantity transmitters and indicators, drain valves, fittings, and connecting lines. The system design shall be in general accordance with MIL-F-38363. Fuel system components not covered by a specific specification shall be in general accordance with MIL-F-8615. The fuel system shall be compatible with JP-4 and JP-5 fuels conforming to MIL-T-5624. The total fuel capacity shall be approximately 5060 pounds (1700 pounds in the wing cells and 3360 pounds in the fuselage cells). This system shall include single-point, pressure fueling provisions.

3.13.10 Propulsion system controls. The propulsion system controls shall be designed to avoid creep due to vibration or unbalanced loads. Push-pull control rods shall be in accordance with MIL-C-7958. Antifriction bearing rod ends shall be provided. Self-locking nuts shall have safety wires or cotter pins. Where threaded fittings are used in the control leads, a double safety device shall be used to prevent the control from parting. All bolts in the propulsion control system shall be self-retaining. A single malfunction or failure in any propulsion control shall not cause the subsystem to fail. Each control shall have sufficient strength and rigidity to withstand operating loads without failure and without excessive deflection. The controls shall be provided with couplings at all connections that may be frequently uncoupled, except for connections that are directly subject to engine vibrations. Quick-disconnect fittings shall be of the approved type in accordance with MIL-F-5442.

3.13.10.1 Power control levers. The collective lever shall have dual twist grip throttles for main engine power control. Twin power levers shall be provided for auxiliary thrust engine control. The thrust engines may also be controlled by the GFE flight control computer.

3.13.11 Starting system. A starter-generator shall be installed on each main engine for starting and DC power generation. A starter shall be installed on each thrust engine for starting. Starter controls shall be provided for the pilot and copilot. In-flight starts may be accomplished.

3.13.12 Main engine drive system. The drive system shall consist of all the components and subsystems required for the transmittal of power from the main engines to the rotors and accessories. The drive system includes the engine combining gearbox, main rotor transmission, input shafting, output shafting, tail rotor and intermediate gearboxes, tail rotor drive shafting and bearing hangers.

3.13.12.1 Main transmission. The main transmission shall be a CH47C forward gearbox rated at 3600 shp at 245 rpm. The transmission shall be mounted inclined 4 degrees forward from the fuselage reference line. The transmission shall accept power inputs from the main engines through a combining gearbox at an output rpm of 6300.

3.13.12.2 Speed reducer gearbox. The engine mounted speed reducer gearbox shall consist of a two-stage reduction unit. The reduction ratio of the gearbox shall be 2.0306 to 1, which reduces the shaft speed of the turbine to 6300 rpm. The gearbox shall have a self-contained pressure lubrication system.

3.13.12.3 Combining gearbox. The combining gearbox shall be attached to each engine through driveshaft assemblies and speed reducer gearboxes. The gearbox shall provide primary power for driving the main transmission with secondary power to drive the tail rotor. Two hydraulic pump drive pads and one AC generator drive pad shall be provided on the gearbox.

3.13.12.4 Intermediate gearbox. The intermediate gearbox shall transmit power from intermediate tailboom shafts to the rear shaft. The gearbox includes an oil level sight gage, an oil filler cap, an indicating type magnetic drain plug, an oil temperature switch, and two similar gear quills. A small oil pump shall be used to provide low pressure lubrication of gear teeth and bearings.

3.13.12.5 Tail rotor gearbox. The tail rotor gearbox shall transmit power from the rear shaft to the tail rotor shaft. The gearbox includes an oil level sight gage, a filler cap, an indicating type magnetic drain plug, and gear quills. A small oil pump shall be used to provide low pressure lubrication of gear teeth and bearings.

3.13.12.6 Freewheeling unit. A sprag-type freewheeling unit shall be installed in the drive system between each speed reducer gearbox and the combining gearbox. The main rotor shall drive the tail rotor when the engines are disengaged. The unit will disconnect the engine(s) and speed-reducer gearbox(es) from the drive system in the event of engine or speed-reducer gearbox failure.

3.14 Secondary power and distribution system.

3.14.1 Electrical power generation and distribution system. Primary electrical power shall be furnished by one 15/20 kVA AC generator mounted on the combining gearbox and two 200 ampere DC starter-generators, one mounted on each main rotor engine. Secondary electrical power shall be provided by one 750 VA standby inverter and one 34 ampere-hour battery.

3.14.1.1 Electrical power supply. An electrical power system shall be provided capable of supplying the rated power as determined by a load analysis prepared in accordance with MIL-E-7016. The simplest bus arrangement for reliable power distribution shall be used. Power characteristics shall conform to MIL-STD-704.

3.14.1.2 Essential bus system. Essential busses shall be provided for all AC and DC loads that are essential for flight under night and instrument conditions. The essential bus system shall provide maximum reliability and least vulnerability to electrical or mechanical damage.

3.14.1.3 Power utilization. Electrical power shall comply with MIL-STD-704. High density harnesses (minimum diameter wire bundles under a protective covering) may be used where reduction in weight and bulk or improved service life can be achieved over that provided by conventional wiring. Wiring and equipment shall be so installed that a minimum of wiring is disturbed when servicing equipment.

3.14.1.4 Equipment installation. Electrical equipment installation shall be in accordance with MIL-E-7080 and MIL-E-25499.

3.14.1.5 Aircraft wiring. Installation of aircraft electrical/electronics wiring shall be in accordance with MIL-W-5088. For weight reduction considerations, MIL-W-5086, MIL-W-81044, or MIL-W-16878 wire may be used.

3.14.1.6 Bonding/grounding. All subsystems and equipment shall be designed to provide adequate facilities for bonding/grounding to the aircraft structure. Bonding methods shall not interfere with installation or removal of the equipment. Bonding/grounding shall conform with the requirements of MIL-B-5087.

3.14.1.7 Lighting. The exterior and interior lighting of the aircraft shall be in accordance with the applicable requirements of MIL-L-6503, MIL-L-6723, and MIL-P-7788. Colors of lights shall conform to the applicable requirements of MIL-C-25050.

3.14.2 Hydraulic power generation and distribution system. The hydraulic system shall consist of two independent hydraulic subsystems. The hydraulic subsystems shall be Type II, 3000 psig type, designed in accordance with MIL-H-5440 for use with MIL-H-83282 hydraulic fluid. Actuating cylinders shall be designed in accordance with MIL-C-5503. The subsystem shall be designed such that a failure in one subsystem shall not adversely affect the other subsystem. Each subsystem shall be fully capable of providing all essential flight control functions when the other subsystem is incapacitated.

3.14.2.1 Primary control subsystem. The primary control subsystem shall provide hydraulic power for the flight control system. The subsystem shall be a closed-centered type and shall consist of a reservoir, a pump, filters, actuators, and interconnecting tubing and fittings. The subsystem shall supply power to one side of the tandem flight control servo-actuators and to the force-feel trim actuators.

3.14.2.2 Utility hydraulic subsystems. The utility hydraulic subsystem shall provide backup hydraulic power for the flight control system. It shall be the same type subsystem as the primary control subsystem. The utility subsystem shall supply power to the second side of the tandem flight control actuators, to the landing gear actuators, and to the brakes. In the event of primary control subsystem failure, all nonflight essential functions shall be isolated by a solenoid operated valve to prevent loss of fluid to the flight control actuators.

3.14.2.3 Hydraulic reservoirs. Nonpneumatic, pressurized reservoirs designed in accordance with MIL-R-8931 shall be used for the hydraulic subsystems. The reservoirs shall be accessible for checkout and servicing.

3.14.2.4 Hydraulic pumps. A variable displacement pressure-compensated pump shall be provided for each subsystem. The pumps shall be designed in general accordance with MIL-P-19692. The pumps shall be mounted on the combining gearbox.

3.14.2.5 Hydraulic actuators. Actuators shall be designed in accordance with MIL-C-5503.

3.14.2.6 Tubing and fittings. Tubing and fittings shall conform to applicable specifications and standards. Detachable type fittings shall be used on components and in those points where frequent disconnection may be necessary.

3.14.2.7 Ground test provisions. Ground test provisions shall be provided. Provisions shall be designed in accordance with MIL-H-5440 and located in an accessible region on one side of the aircraft.

3.15 Avionics subsystem. The avionics subsystem shall be designed to meet the requirements of MIL-E-5400 to the extent required for a research aircraft. Installation of the equipment shall be in accordance with MIL-I-8700 and applicable Government or equipment manufacturer's specifications.

3.15.1 Installed equipment. The avionic equipment to be installed shall consist of the communication and navigation equipment listed below:

Communication subsystems

Two	C-6533()/ARC Intercommunication Controls
One	AN/ARC-115 VHF-AM Radio Set
One	AN/ARC-116 UHF-AM Radio Set

Navigation subsystem

One	AN/APX-72 IFF Transponder
One	VOR/DME
One	Instrument Landing System
One	AN/ASN-43 Gyromagnetic Compass Set

3.16 Aircraft handling and servicing provisions.

3.16.1 Towing provisions. Towing provisions shall be provided in accordance with MIL-STD-805.

3.16.2 Jacking provisions. Provisions for jacking shall be made in accordance with MIL-STD-809. Wing jacking provisions shall be provided in an appropriate position for calibrating the wing balance.

3.16.3 Tie-down provisions. Tie-down provisions shall be in accordance with MIL-T-81259.

3.16.4 Hoisting provisions. Provisions shall be made for hoisting the aircraft from the top of the mast.

3.16.5 Leveling provisions. Provisions for measuring and leveling the aircraft for weighing and structural alignment shall be in accordance with MIL-M-6756.

3.16.6 Servicing. The engines, main transmission, speed reducer gearbox, combining gearbox, intermediate gearbox, tail rotor gearbox, and the freewheeling unit lubricating system will be easily serviceable. The fuel system shall be a single-point pressure fueling type.

3.17 Instrument subsystem. Flight navigation, and power plant instruments for use by the pilot and copilot shall be plainly visible from their stations with minimum practical deviation from normal position and line of vision when looking out and forward along the flight path. The instruments shall be located such that parallax error is negligible. The pilot's instrument panel management shall be substantially in accordance with MIL-STD-250 and MS33785. The copilot's instrument panel shall contain only the essential instruments for flight under IFR or visual conditions.

3.17.1 Installation. The instrument panels shall be shock mounted. All instruments shall be removable from the front of the panel. Suitable range markings shall be provided on the instruments in accordance with TM55-6600-200-20. Instrument lighting shall be in accordance with MIL-L-25467.

3.17.2 Instruments. The following instruments shall be provided on the instrument panels:

Pilot's Panel

1 Airspeed

1 Altimeter

1 Attitude

1 Rate of climb

1 Turn and bank

1 Standby compass

1 Clock

1 ILS

1 Gyromagnetic compass

1 VOR/DME

1 Outside air temperature

3.17.2 (Continued)

1	Triple tachometer [main engines (N_1) and main rotor]
2	Main engine oil pressure and temperature
2	Main engine EGT
2	Auxiliary engine pressure ratio
2	Auxiliary engine oil pressure and temperature
2	Auxiliary engine rpm
2	Auxiliary engine EGT
1	Sensitive tachometer
1	Transmission oil pressure and temperature
1	Combining gearbox oil pressure and temperature
1	Torquemeter
1	Ammeter
1	Voltmeter
1	Fuel pressure
1	Fuel quantity
1	Upper flap position
1	Lower flap position (dual indicator needles)
1	Wing incidence
1	Tail rotor collective washout
1	Fore and aft cyclic washout
1	Main rotor collective washout

3.17.2 (Continued)

Copilot's Panel

1 Airspeed

1 Altimeter

1 Attitude

1 Rate of climb

1 Turn and bank

1 Standby compass

1 Clock

1 ILS

1 Gyromagnetic compass

1 VOR/DME

1 Triple tachometer

2 Main engine oil pressure and temperature

2 Main engine EGT

2 Auxiliary engine pressure ratio

2 Auxiliary engine oil pressure and temperature

2 Auxiliary engine rpm

2 Auxiliary engine EGT

1 Torquemeter

1 Ammeter

1 Voltmeter

1 Fuel quantity

1 Lower flap position (dual indicator needles)

1 Wing incidence

3.17.2 (Continued)

1 Tail rotor washout

1 Fore and aft cyclic washout

1 Collective washout

3.17.3 Caution/warning/advisory signals. Caution/warning/advisory signals shall be in accordance with MIL-S-38039 and MIL-STD-411. Word warning caution panels shall be provided for the pilot and copilot.

3.18 Environmental control subsystem. The environmental control system shall consist of a cockpit environmental control system, cockpit ventilation system, defrost/defogging system, and equipment compartment cooling system.

3.18.1 Cockpit environment control. Cockpit environment control shall be provided for all crew stations. The system shall be capable of providing approximately 65°F cockpit heating temperature at 0°F OAT to 15,000 feet altitude and 85°F cockpit cooling temperature at 110°F OAT at sea level.

3.18.2 Cockpit ventilation. Cockpit ventilation shall provide forced air when the cockpit environment control system is not operating. The ventilation system shall distribute air uniformly over the crew compartment. The source of air shall be outside of the aircraft, free from possible contamination from the aircraft exhaust system and hydraulic and fuel lines.

3.18.3 Transparent areas. Defrost/defogging shall be accomplished by blowing bleed air over the canopy.

3.18.4 Equipment compartment cooling. Cooling shall be provided as necessary to insure that the maximum continuous ambient temperatures specified for the various installed equipments are not exceeded in flight.

3.19 Data acquisition system. A data acquisition system compatible with the NASA recording system shall be provided. The system shall provide a means to measure the following data:

a. Main rotor (excluding blades)

(1) Rotor rpm and azimuth

(2) Hub moment

(3) Torque

3.19 (Continued)

(4) Lift

(5) Drag

(6) Side force

(7) Lateral and longitudinal cyclic stick position

(8) Collective stick position

(9) Boost tube positions

(10) SCAS input

b. Main rotor blades

(1) Feathering

(2) Flapping

(3) Chordwise pressure distribution (upper and lower surface at eleven chord stations at five radial stations)

(4) Flow direction (upper and lower surfaces at three chord stations at five radial stations)

(5) Angle of attack (five radial stations)

(6) Chordwise bending moment (eleven radial stations)

(7) Beamwise bending moment (eleven radial stations)

(8) Chordwise acceleration (eleven radial stations)

(9) Beamwise acceleration (eleven radial stations)

c. Other measurements

(1) Wing incidence

(2) Wing chordwise and beamwise forces

(3) Tail rotor shaftwise force

(4) Stabilator beamwise force

3.19 (Continued)

(5) Auxiliary engine thrust

(6) Aircraft state

(a) Altitude

(b) Airspeed

(c) Fuselage angle of attack

(d) Fuselage sideslip angle

(e) Three-axes angular accelerations and rates

(f) Three-axes linear accelerations

(g) Roll and pitch attitude

4. QUALITY ASSURANCE PROVISIONS

4.1 Aircraft qualification tests. Aircraft qualification tests shall be conducted in accordance with the aircraft development plan outlined in BHC Report 646-099-003.

4.2 Flight testing. Flight testing shall be conducted in accordance with the aircraft development plan outlined in BHC Report 646-099-003.

5. PREPARATION FOR DELIVERY

5.1 The aircraft shall be prepared for delivery as required by the contract.

6. NOTES

6.1 Intended use. This specification is intended to provide a description of the aircraft to be developed.

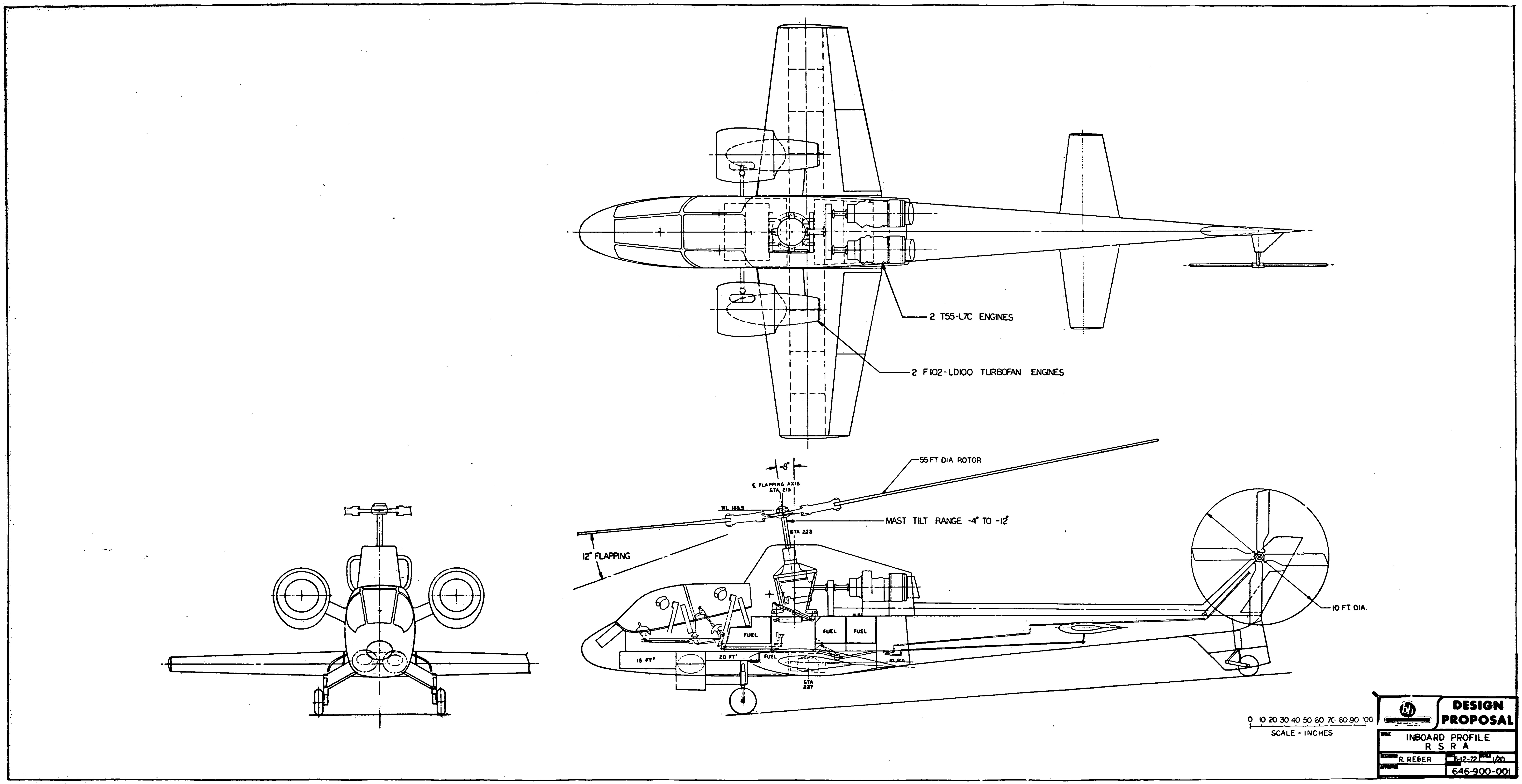
6.2 Specification format. This specification was written using MIL-STD-832 as a guide.

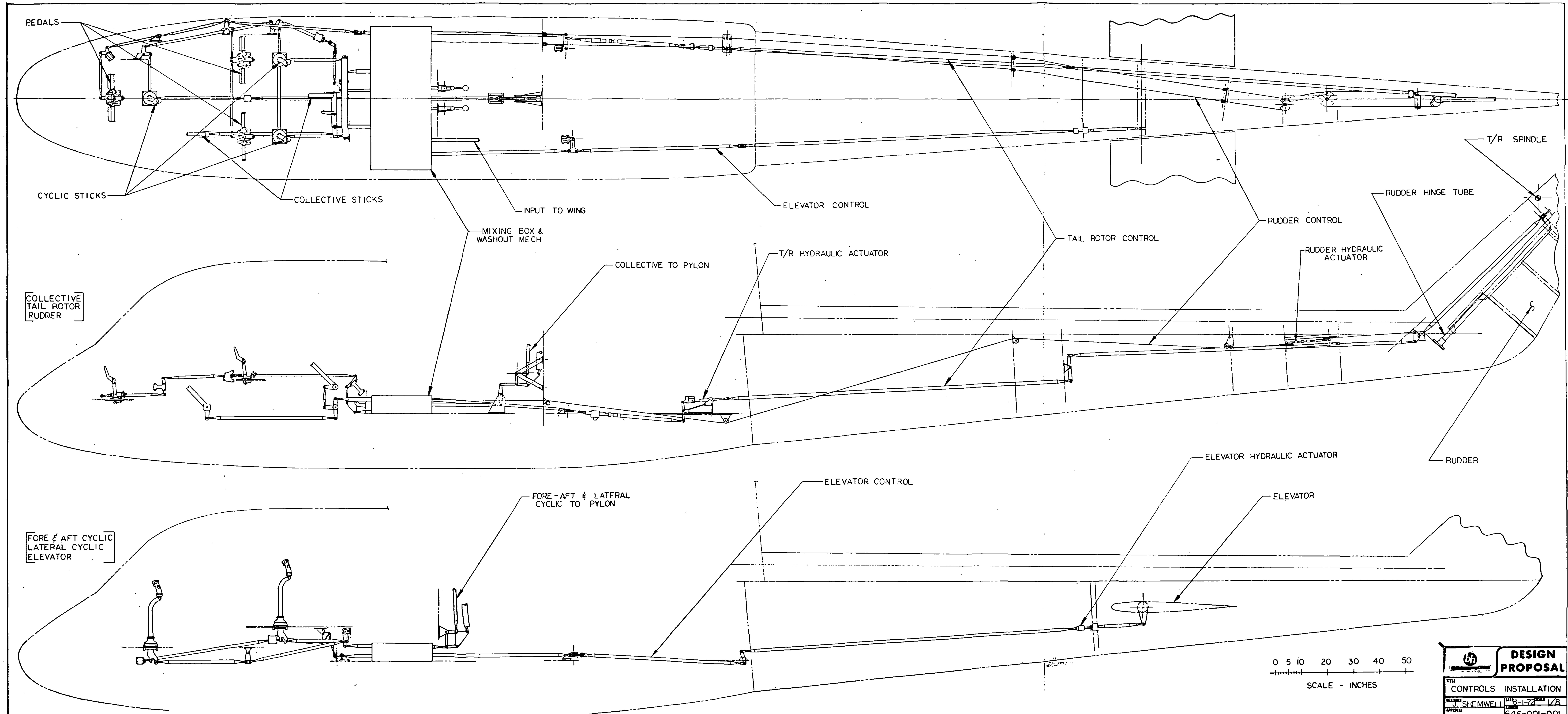
6.3 Deviations. The contractor may take exceptions/deviations to the military documents listed herein, when these exceptions can increase performance, safety, or decrease cost.

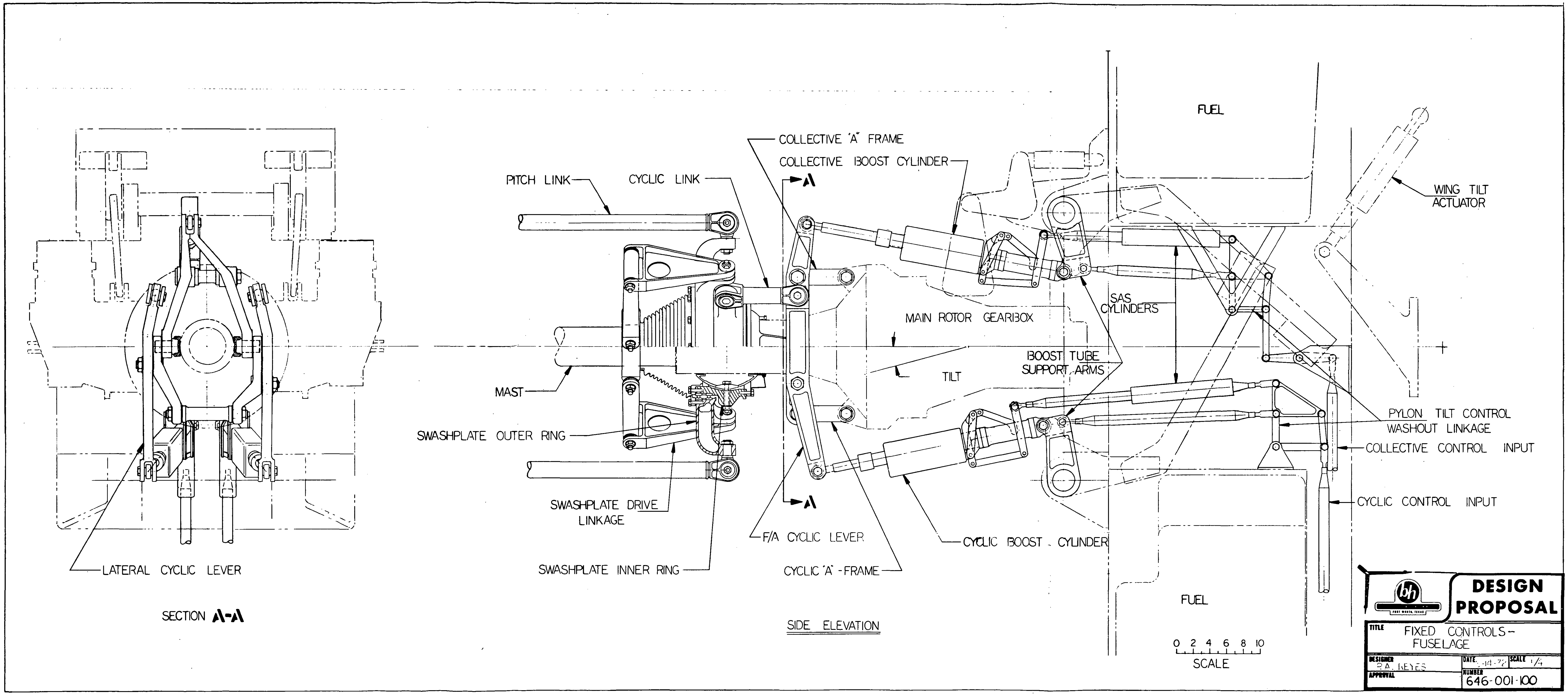
APPENDIX 2

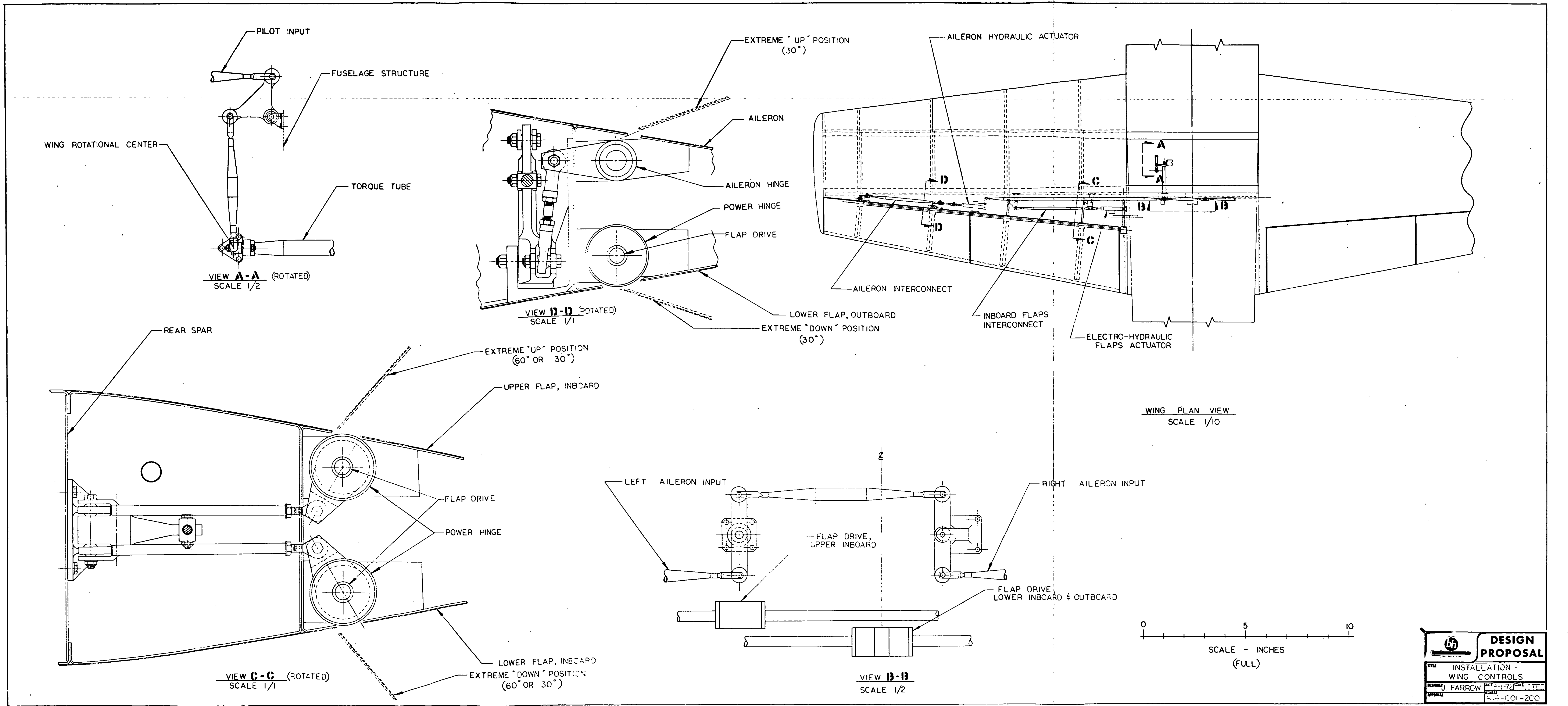
DRAWINGS

LIST OF DRAWINGS

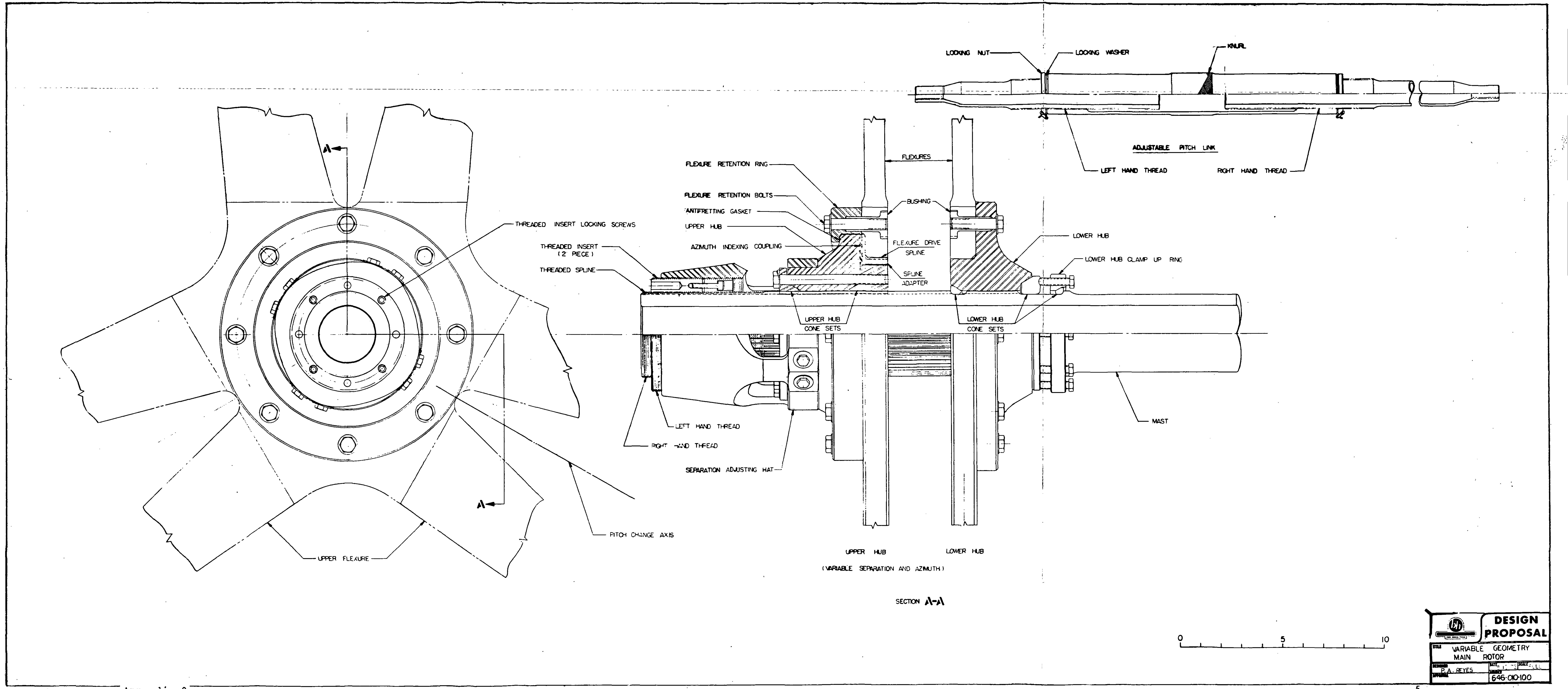

2 T55-L7C ENGINES
2 F102-LD100 TURBOFAN ENGINES
55 FT DIA ROTOR
-8°
℄ FLAPPING AXIS
STA 213
WL 183.5
MAST TILT RANGE -4° TO -12°
STA 223
12° FLAPPING
FUEL
FUEL
FUEL
15 FT³
20 FT³
FUEL
STA
237
10 FT. DIA.
0 10 20 30 40 50 60 70 80 90 100
SCALE - INCHES
DESIGN
PROPOSAL
INBOARD PROFILE
R S R A
R. REBER
5-12-72
1/20
646-900-001

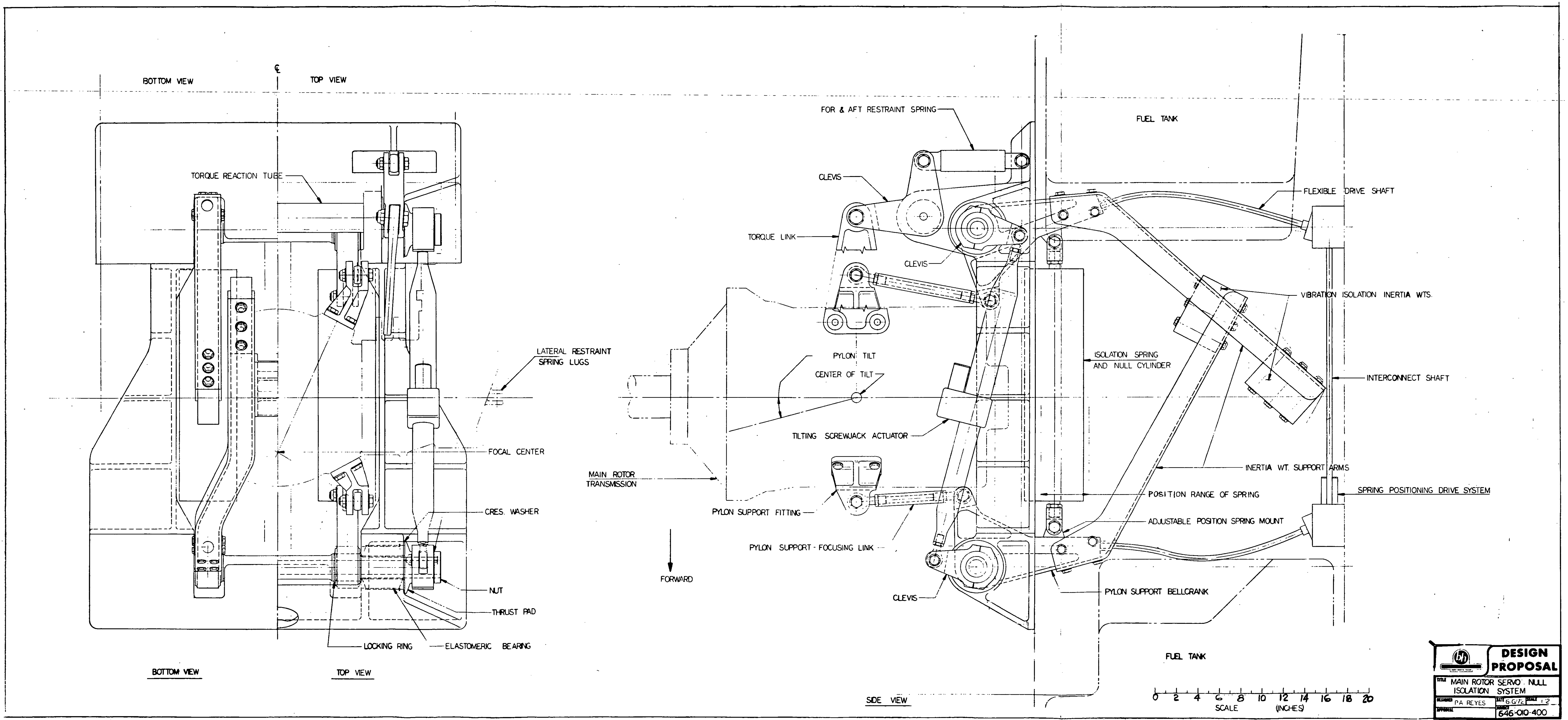

PEDALS
CYCLIC STICKS
COLLECTIVE STICKS
INPUT TO WING
MIXING BOX & WASHOUT MECH
ELEVATOR CONTROL
TAIL ROTOR CONTROL
RUDDER CONTROL
RUDDER HINGE TUBE
T/R SPINDLE
COLLECTIVE TO PYLON
T/R HYDRAULIC ACTUATOR
RUDDER HYDRAULIC ACTUATOR
COLLECTIVE
TAIL ROTOR
RUDDER
RUDDER
FORE & AFT CYCLIC
LATERAL CYCLIC
ELEVATOR
FORE-AFT & LATERAL CYCLIC TO PYLON
ELEVATOR CONTROL
ELEVATOR HYDRAULIC ACTUATOR
ELEVATOR
0 5 10 20 30 40 50
SCALE - INCHES
DESIGN PROPOSAL
CONTROLS INSTALLATION
J. SHEMWELL
8-1-72
1/8
646-001-001

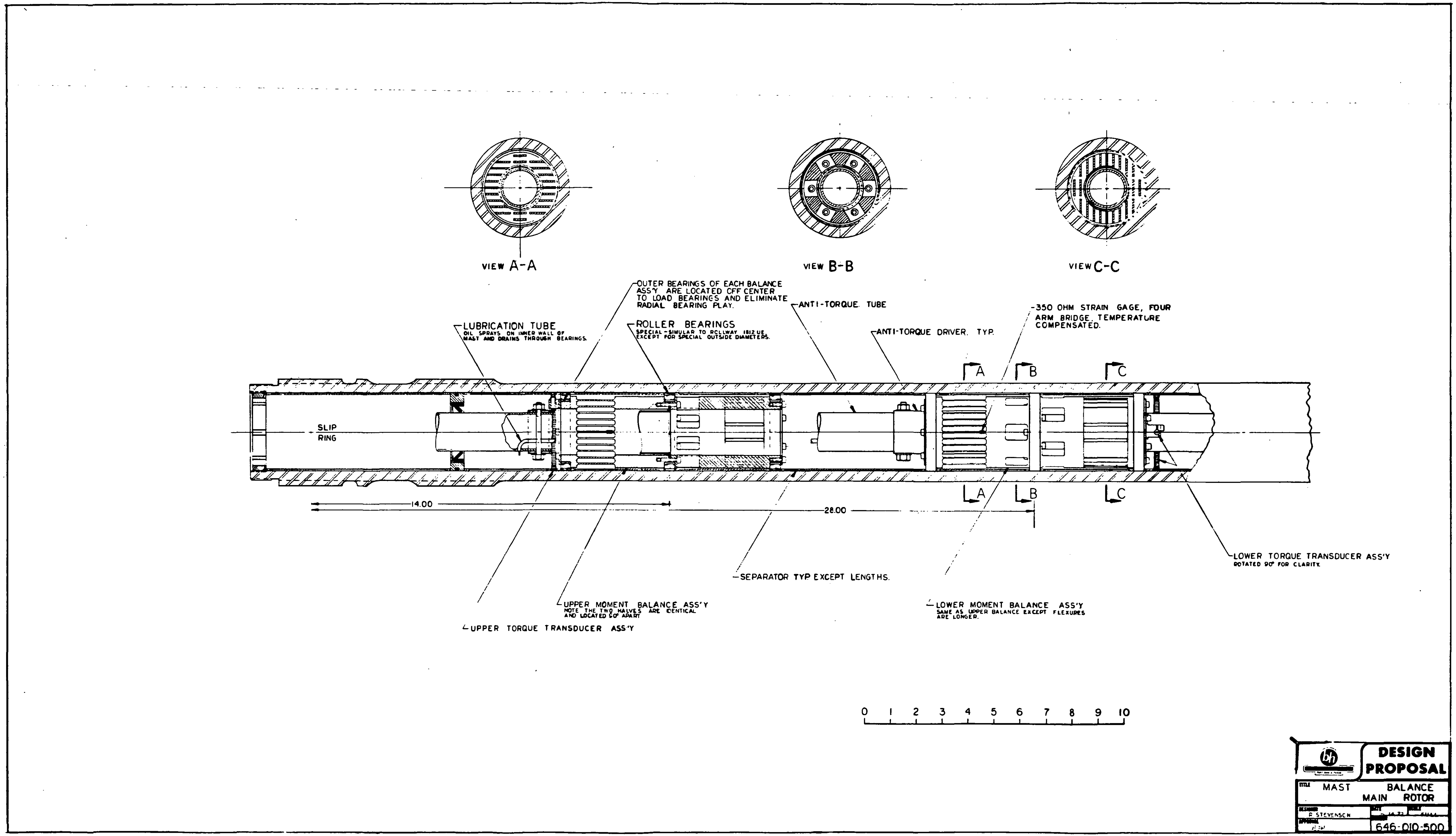

FUEL
COLLECTIVE 'A' FRAME
COLLECTIVE BOOST CYLINDER
PITCH LINK
CYCLIC LINK
A
WING TILT ACTUATOR
SAS CYLINDERS
MAIN ROTOR GEARBOX
BOOST TUBE SUPPORT ARMS
TILT
MAST
SWASHPLATE OUTER RING
PYLON TILT CONTROL WASHOUT LINKAGE
COLLECTIVE CONTROL INPUT
CYCLIC CONTROL INPUT
SWASHPLATE DRIVE LINKAGE
F/A CYCLIC LEVER
CYCLIC BOOST CYLINDER
LATERAL CYCLIC LEVER
SWASHPLATE INNER RING
CYCLIC 'A' -FRAME
FUEL
DESIGN PROPOSAL
SECTION A-A
SIDE ELEVATION
0 2 4 6 8 10
SCALE
TITLE FIXED CONTROLS - FUSELAGE
DESIGNER P.A. KEYES
DATE 5-14-72
SCALE 1/4
APPROVAL
NUMBER 646-001-100

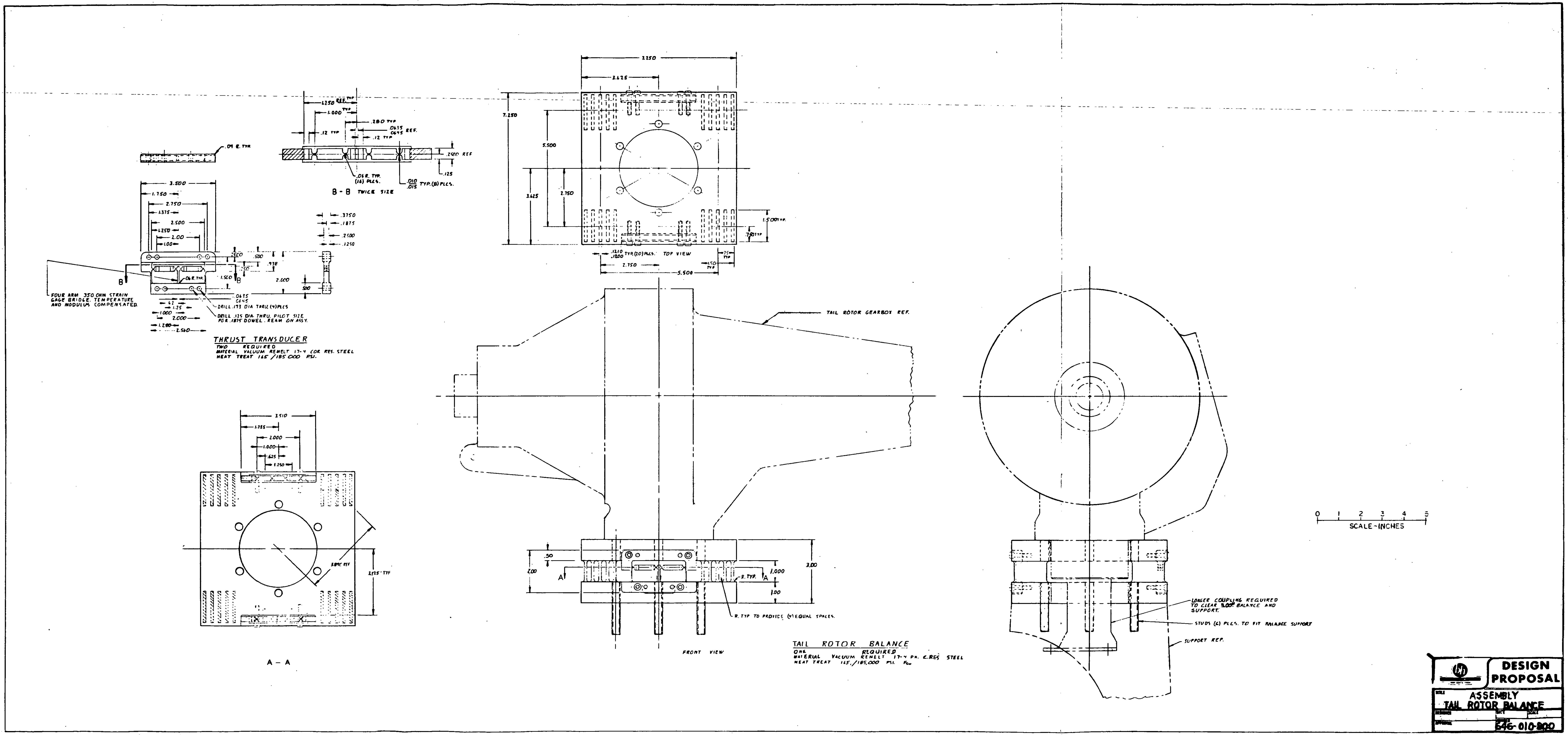

PILOT INPUT
FUSELAGE STRUCTURE
WING ROTATIONAL CENTER
TORQUE TUBE
VIEW A-A (ROTATED)
SCALE 1/2
EXTREME "UP" POSITION
(30°)
AILERON
AILERON HINGE
POWER HINGE
FLAP DRIVE
VIEW D-D (ROTATED)
SCALE 1/1
LOWER FLAP, OUTBOARD
EXTREME "DOWN" POSITION
(30°)
AILERON HYDRAULIC ACTUATOR
AILERON INTERCONNECT
INBOARD FLAPS
INTERCONNECT
ELECTRO-HYDRAULIC
FLAPS ACTUATOR
WING PLAN VIEW
SCALE 1/10
REAR SPAR
EXTREME "UP" POSITION
(60° OR 30°)
UPPER FLAP, INBOARD
FLAP DRIVE
POWER HINGE
LOWER FLAP, INBOARD
EXTREME "DOWN" POSITION
(60° OR 30°)
VIEW C-C (ROTATED)
SCALE 1/1
LEFT AILERON INPUT
RIGHT AILERON INPUT
FLAP DRIVE,
UPPER INBOARD
FLAP DRIVE
LOWER INBOARD & OUTBOARD
VIEW B-B
SCALE 1/2
0
5
10
SCALE - INCHES
(FULL)
DESIGN
PROPOSAL
INSTALLATION -
WING CONTROLS
J. FARROW

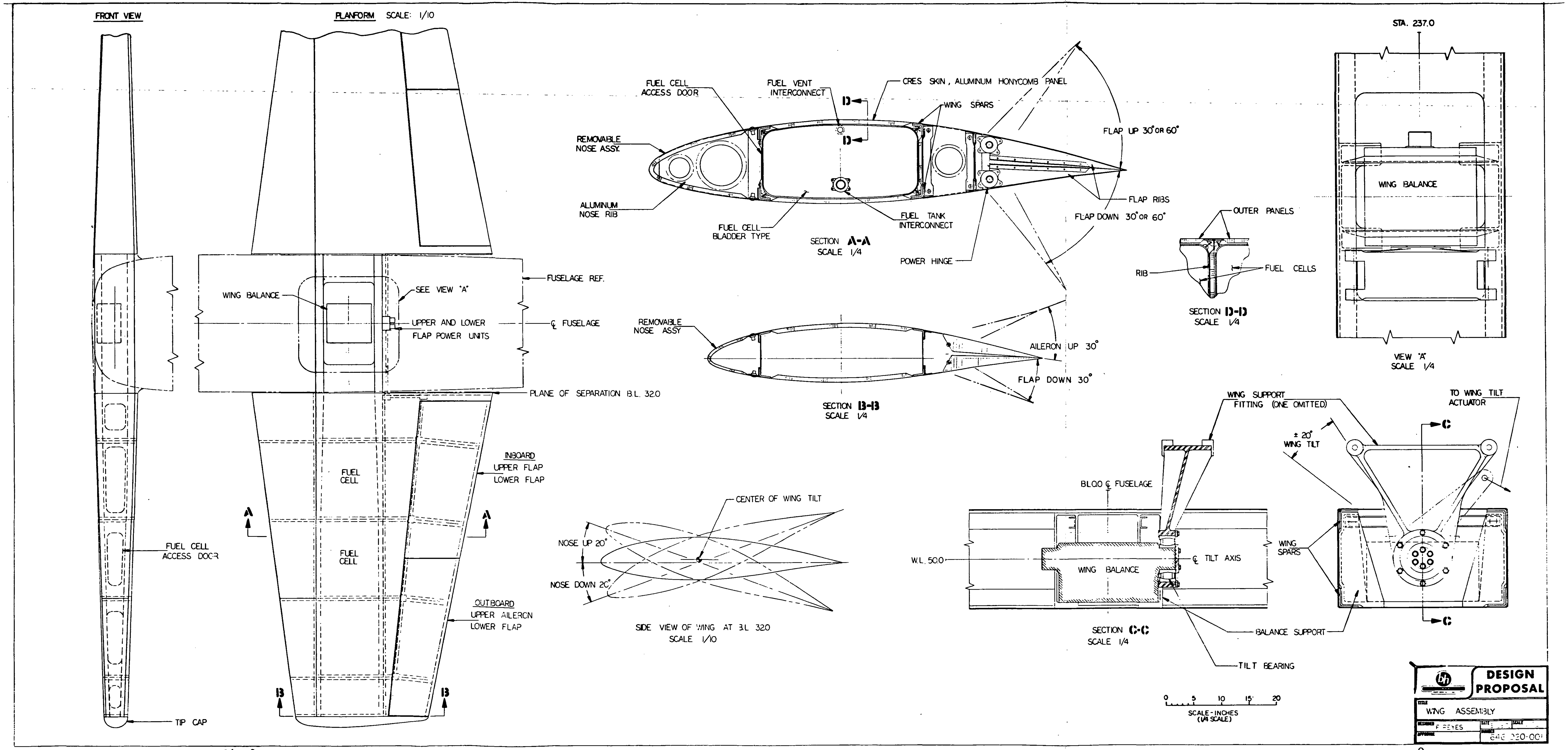
Appendix 2


LOCKING NUT
LOCKING WASHER
KNURL
ADJUSTABLE PITCH LINK
LEFT HAND THREAD
RIGHT HAND THREAD
FLEXURES
FLEXURE RETENTION RING
FLEXURE RETENTION BOLTS
ANTIFRETTING GASKET
UPPER HUB
AZIMUTH INDEXING COUPLING
BUSHING
FLEXURE DRIVE SPLINE
SPLINE ADAPTER
LOWER HUB
LOWER HUB CLAMP UP RING
THREADED INSERT LOCKING SCREWS
THREADED INSERT (2 PIECE)
THREADED SPLINE
UPPER HUB CONE SETS
LOWER HUB CONE SETS
A
A
LEFT HAND THREAD
RIGHT HAND THREAD
SEPARATION ADJUSTING HAT
MAST
PITCH CHANGE AXIS
UPPER FLEXURE
UPPER HUB
LOWER HUB
(VARIABLE SEPARATION AND AZIMUTH)
SECTION A-A
0
5
10
DESIGN PROPOSAL
VARIABLE GEOMETRY MAIN ROTOR
P.A. REYES
646-010-100

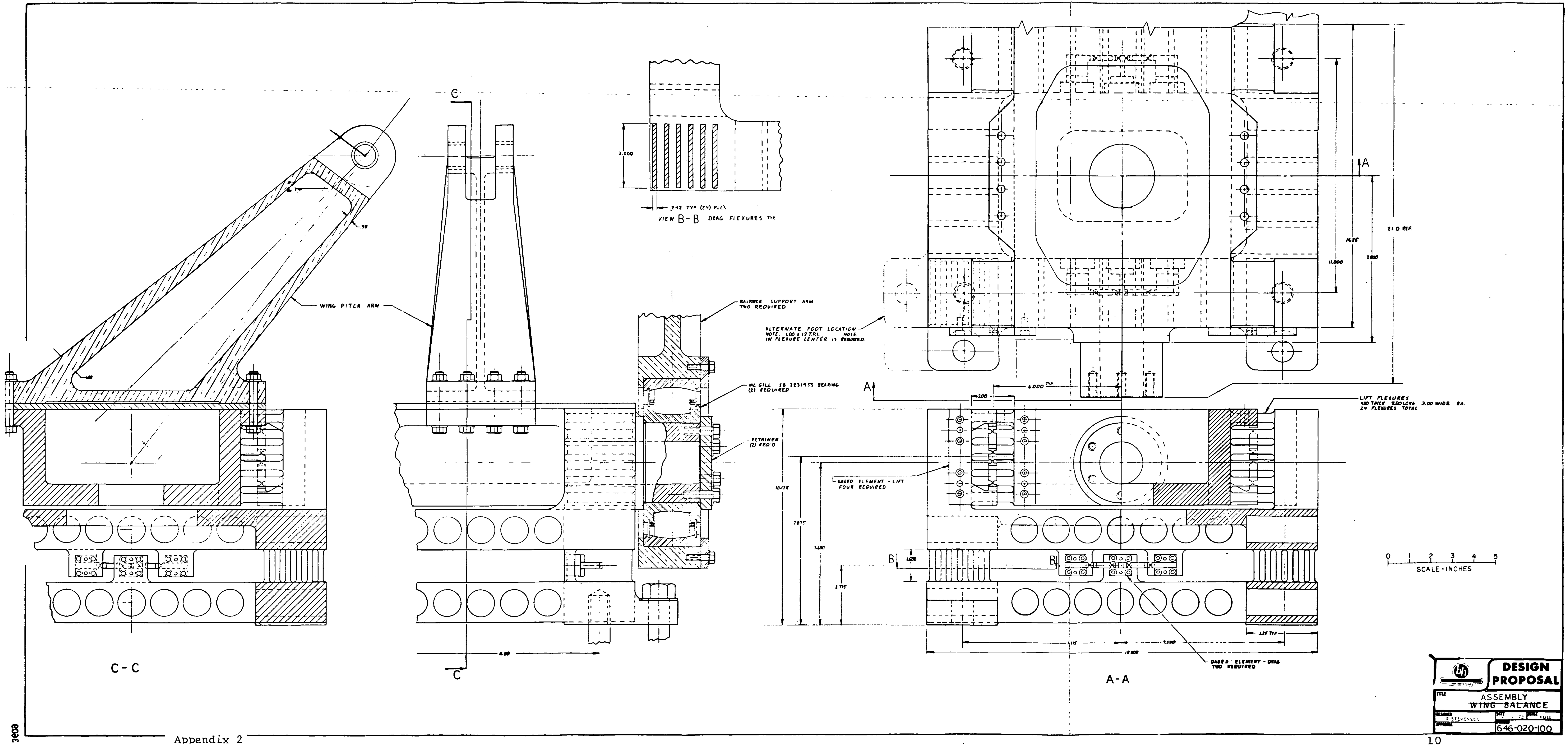

BOTTOM VIEW
TOP VIEW
TORQUE REACTION TUBE
LATERAL RESTRAINT SPRING LUGS
FOCAL CENTER
CRES. WASHER
NUT
THRUST PAD
LOCKING RING
ELASTOMERIC BEARING
BOTTOM VIEW
TOP VIEW
FOR & AFT RESTRAINT SPRING
CLEVIS
TORQUE LINK
CLEVIS
FUEL TANK
FLEXIBLE DRIVE SHAFT
VIBRATION ISOLATION INERTIA WTS.
PYLON TILT
CENTER OF TILT
ISOLATION SPRING AND NULL CYLINDER
INTERCONNECT SHAFT
TILTING SCREWJACK ACTUATOR
MAIN ROTOR TRANSMISSION
INERTIA WT. SUPPORT ARMS
SPRING POSITIONING DRIVE SYSTEM
POSITION RANGE OF SPRING
PYLON SUPPORT FITTING
ADJUSTABLE POSITION SPRING MOUNT
PYLON SUPPORT - FOCUSING LINK
FORWARD
CLEVIS
PYLON SUPPORT BELLCRANK
FUEL TANK
SIDE VIEW
0 2 4 6 8 10 12 14 16 18 20
SCALE (INCHES)
DESIGN PROPOSAL
MAIN ROTOR SERVO NULL ISOLATION SYSTEM
P.A. REYES
646-010-400

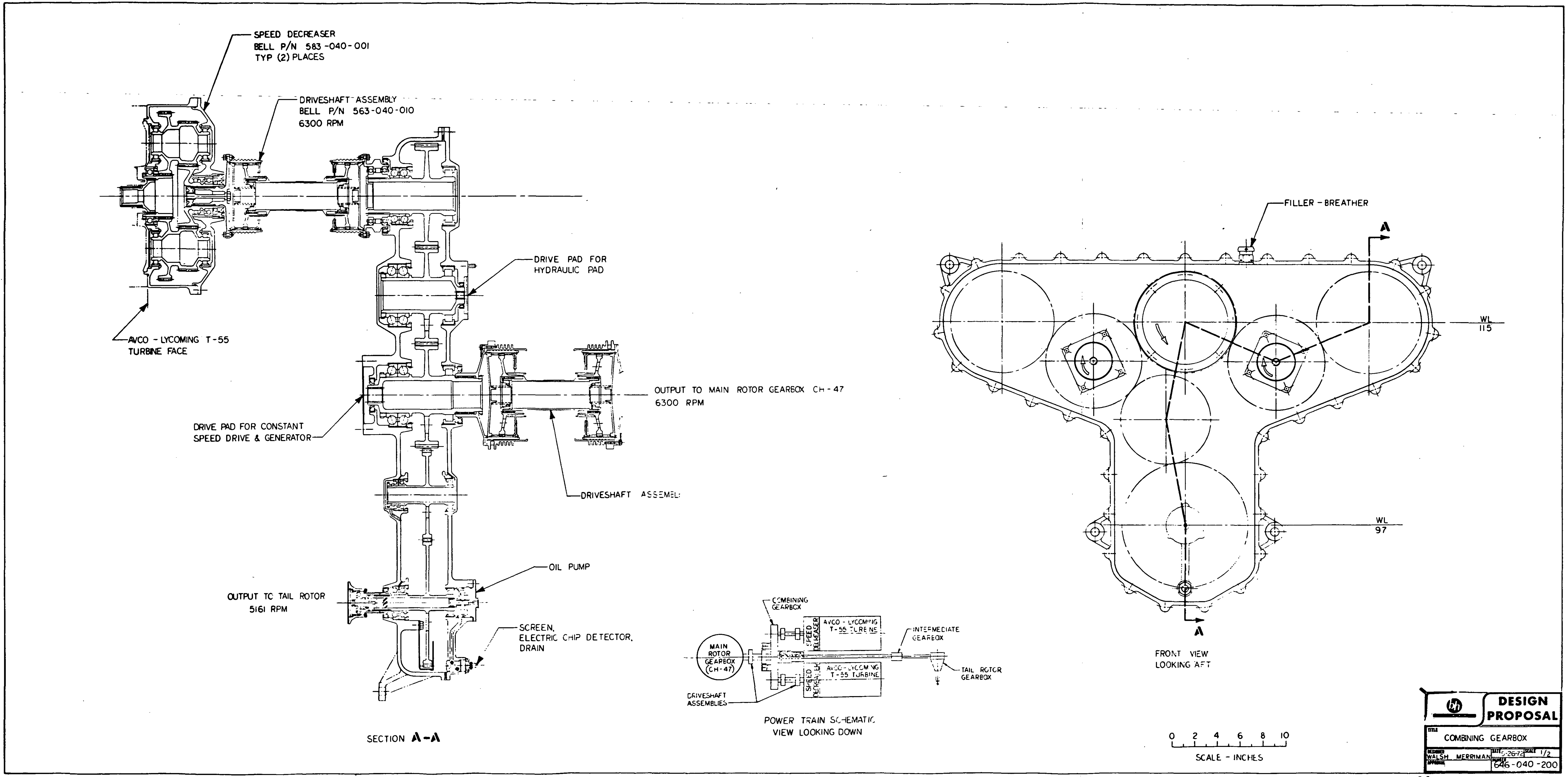

VIEW A-A
VIEW B-B
VIEW C-C
OUTER BEARINGS OF EACH BALANCE ASS'Y ARE LOCATED CFF CENTER TO LOAD BEARINGS AND ELIMINATE RADIAL BEARING PLAY.
ANTI-TORQUE TUBE
350 OHM STRAIN GAGE, FOUR ARM BRIDGE, TEMPERATURE COMPENSATED.
LUBRICATION TUBE
OIL SPRAYS ON INNER WALL OF MAST AND DRAINS THROUGH BEARINGS.
ROLLER BEARINGS
SPECIAL - SIMULAR TO RCLLWAY 1812 UE. EXCEPT FOR SPECIAL OUTSIDE DIAMETERS.
ANTI-TORQUE DRIVER. TYP.
A
B
C
SLIP RING
14.00
28.00
SEPARATOR TYP EXCEPT LENGTHS.
LOWER TORQUE TRANSDUCER ASS'Y
ROTATED 90° FOR CLARITY.
UPPER MOMENT BALANCE ASS'Y
NOTE THE TWO HALVES ARE IDENTICAL AND LOCATED 90° APART
LOWER MOMENT BALANCE ASS'Y
SAME AS UPPER BALANCE EXCEPT FLEXURES ARE LONGER.
UPPER TORQUE TRANSDUCER ASS'Y
0 1 2 3 4 5 6 7 8 9 10
DESIGN PROPOSAL
TITLE MAST BALANCE MAIN ROTOR
R STEVENSON
646-010-500

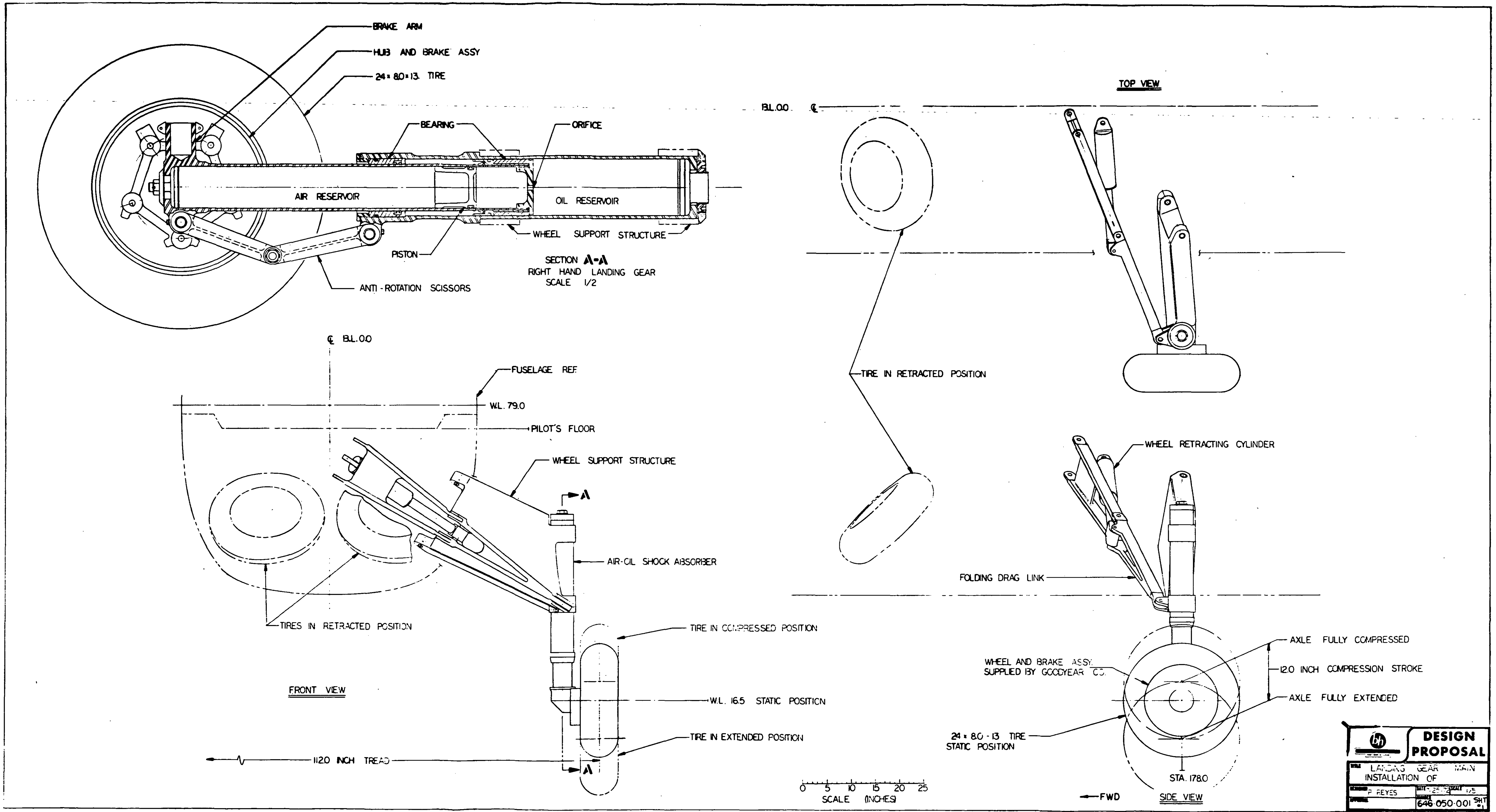

THRUST TRANSDUCER
TWO REQUIRED
MATERIAL VACUUM REMELT 17-4 COR RES. STEEL
HEAT TREAT 165 /185 000 PSI.
FOUR ARM 350 OHM STRAIN GAGE BRIDGE. TEMPERATURE AND MODULUS COMPENSATED.
DRILL .173 DIA THRU, (4) PLCS
DRILL .125 DIA THRU, PILOT SIZE FOR .1875 DOWEL, REAM ON ASSY.
B - B TWICE SIZE
TOP VIEW
TAIL ROTOR GEARBOX REF.
FRONT VIEW
A - A
TAIL ROTOR BALANCE
ONE REQUIRED
MATERIAL VACUUM REMELT 17-4 PH. C.RES STEEL
HEAT TREAT 165./185,000 PSI.
R. TYP TO PROVIDE (4) EQUAL SPACES.
LONGER COUPLING REQUIRED TO CLEAR 3.00 BALANCE AND SUPPORT.
STUDS (6) PLCS. TO FIT BALANCE SUPPORT
SUPPORT REF.
SCALE-INCHES
DESIGN PROPOSAL
ASSEMBLY
TAIL ROTOR BALANCE
546-010-800



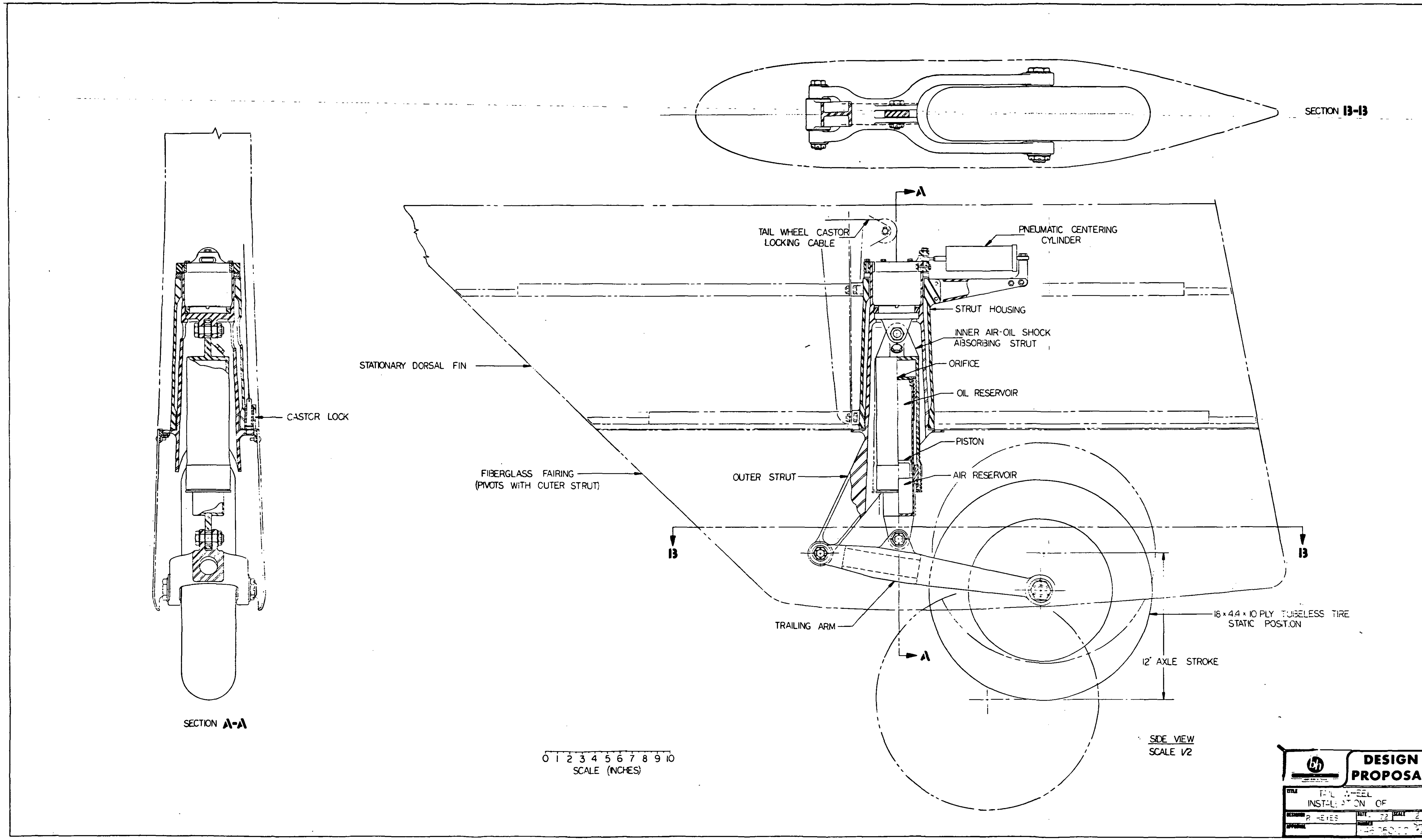
FRONT VIEW
PLANFORM SCALE: 1/10
FUEL CELL ACCESS DOOR
FUEL VENT INTERCONNECT
CRES SKIN, ALUMINUM HONYCOMB PANEL
WING SPARS
REMOVABLE NOSE ASSY.
ALUMINUM NOSE RIB
FUEL CELL BLADDER TYPE
FUEL TANK INTERCONNECT
FLAP UP 30° OR 60°
FLAP RIBS
FLAP DOWN 30° OR 60°
POWER HINGE
SECTION A-A
SCALE 1/4
STA. 237.0
WING BALANCE
OUTER PANELS
RIB
FUEL CELLS
SECTION D-D
SCALE 1/4
VIEW "A"
SCALE 1/4
FUSELAGE REF.
SEE VIEW "A"
UPPER AND LOWER FLAP POWER UNITS
℄ FUSELAGE
REMOVABLE NOSE ASSY
AILERON UP 30°
FLAP DOWN 30°
SECTION B-B
SCALE 1/4
PLANE OF SEPARATION B.L. 32.0
INBOARD
UPPER FLAP
LOWER FLAP
FUEL CELL
FUEL CELL
FUEL CELL ACCESS DOOR
OUTBOARD
UPPER AILERON
LOWER FLAP
TIP CAP
CENTER OF WING TILT
NOSE UP 20°
NOSE DOWN 20°
SIDE VIEW OF WING AT B.L. 32.0
SCALE 1/10
WING SUPPORT FITTING (ONE OMITTED)
TO WING TILT ACTUATOR
± 20° WING TILT
BL 0.0 ℄ FUSELAGE
W.L. 50.0
℄ TILT AXIS
WING SPARS
SECTION C-C
SCALE 1/4
BALANCE SUPPORT
TILT BEARING
0 5 10 15 20
SCALE-INCHES (1/4 SCALE)
DESIGN PROPOSAL
WING ASSEMBLY

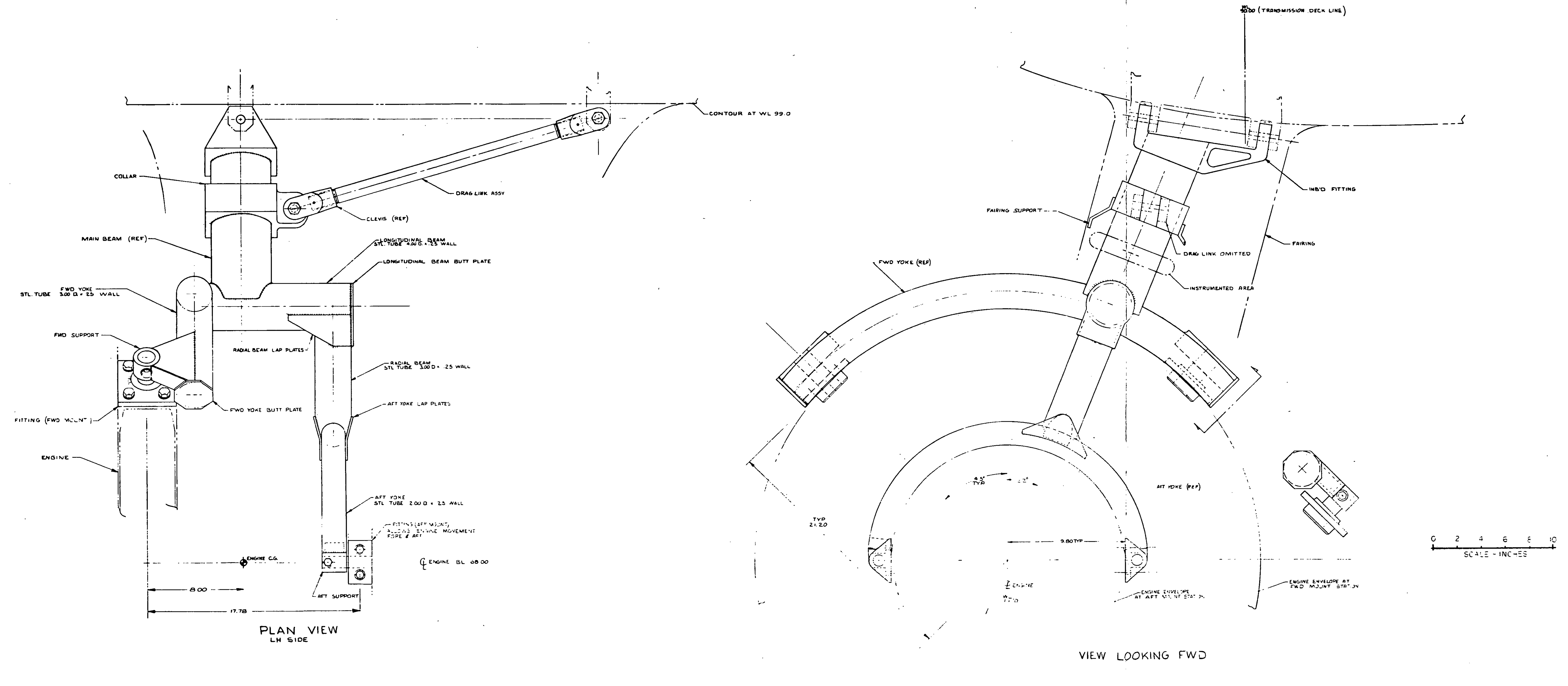

C - C
C
C
WING PITCH ARM
VIEW B - B DRAG FLEXURES TYP.
.242 TYP (24) PLC'S
3.000
BALANCE SUPPORT ARM
TWO REQUIRED
MC GILL SB 22319 SS BEARING
(2) REQUIRED
RETAINER
(2) REQ'D
ALTERNATE FOOT LOCATION
NOTE. 1.00 X 12 T.P.I. HOLE
IN FLEXURE CENTER IS REQUIRED
A
A
21.0 REF.
LIFT FLEXURES
.400 THICK 2.00 LONG 3.00 WIDE EA.
24 FLEXURES TOTAL
GAGED ELEMENT - LIFT
FOUR REQUIRED
GAGED ELEMENT - DRAG
TWO REQUIRED
6.000 TYP
B
B
A - A
0 1 2 3 4 5
SCALE - INCHES
DESIGN PROPOSAL
TITLE
ASSEMBLY
~~WING BALANCE~~
F. STEVENSON
FULL
646-020-100


SPEED DECREASER
BELL P/N 583-040-001
TYP (2) PLACES
DRIVESHAFT ASSEMBLY
BELL P/N 563-040-010
6300 RPM
DRIVE PAD FOR
HYDRAULIC PAD
AVCO - LYCOMING T-55
TURBINE FACE
OUTPUT TO MAIN ROTOR GEARBOX CH-47
6300 RPM
DRIVE PAD FOR CONSTANT
SPEED DRIVE & GENERATOR
DRIVESHAFT ASSEMBL
OIL PUMP
OUTPUT TO TAIL ROTOR
5161 RPM
SCREEN,
ELECTRIC CHIP DETECTOR,
DRAIN
SECTION A-A
COMBINING
GEARBOX
AVCO - LYCOMING
T-55 TURBINE
SPEED
DECREASER
MAIN
ROTOR
GEARBOX
(CH-47)
INTERMEDIATE
GEARBOX
TAIL ROTOR
GEARBOX
DRIVESHAFT
ASSEMBLIES
POWER TRAIN SCHEMATIC
VIEW LOOKING DOWN
FILLER - BREATHER
A
WL
115
WL
97
FRONT VIEW
LOOKING AFT
0 2 4 6 8 10
SCALE - INCHES
DESIGN
PROPOSAL
COMBINING GEARBOX
WALSH MERRIMAN
5-26-72
1/2
646-040-200



BRAKE ARM
HUB AND BRAKE ASSY
24 x 8.0 x 13 TIRE
BEARING
ORIFICE
AIR RESERVOIR
OIL RESERVOIR
WHEEL SUPPORT STRUCTURE
PISTON
ANTI-ROTATION SCISSORS
SECTION A-A
RIGHT HAND LANDING GEAR
SCALE 1/2
B.L.0.0
TOP VIEW
TIRE IN RETRACTED POSITION
FUSELAGE REF
W.L. 79.0
PILOT'S FLOOR
WHEEL SUPPORT STRUCTURE
A
AIR-OIL SHOCK ABSORBER
TIRES IN RETRACTED POSITION
TIRE IN COMPRESSED POSITION
FRONT VIEW
W.L. 16.5 STATIC POSITION
TIRE IN EXTENDED POSITION
112.0 INCH TREAD
A
WHEEL RETRACTING CYLINDER
FOLDING DRAG LINK
AXLE FULLY COMPRESSED
WHEEL AND BRAKE ASSY
SUPPLIED BY GOODYEAR CO.
12.0 INCH COMPRESSION STROKE
AXLE FULLY EXTENDED
24 x 8.0 - 13 TIRE
STATIC POSITION
STA. 178.0
0 5 10 15 20 25
SCALE (INCHES)
FWD
SIDE VIEW
DESIGN
PROPOSAL
LANDING GEAR MAIN
INSTALLATION OF
P. REYES
646-050-001

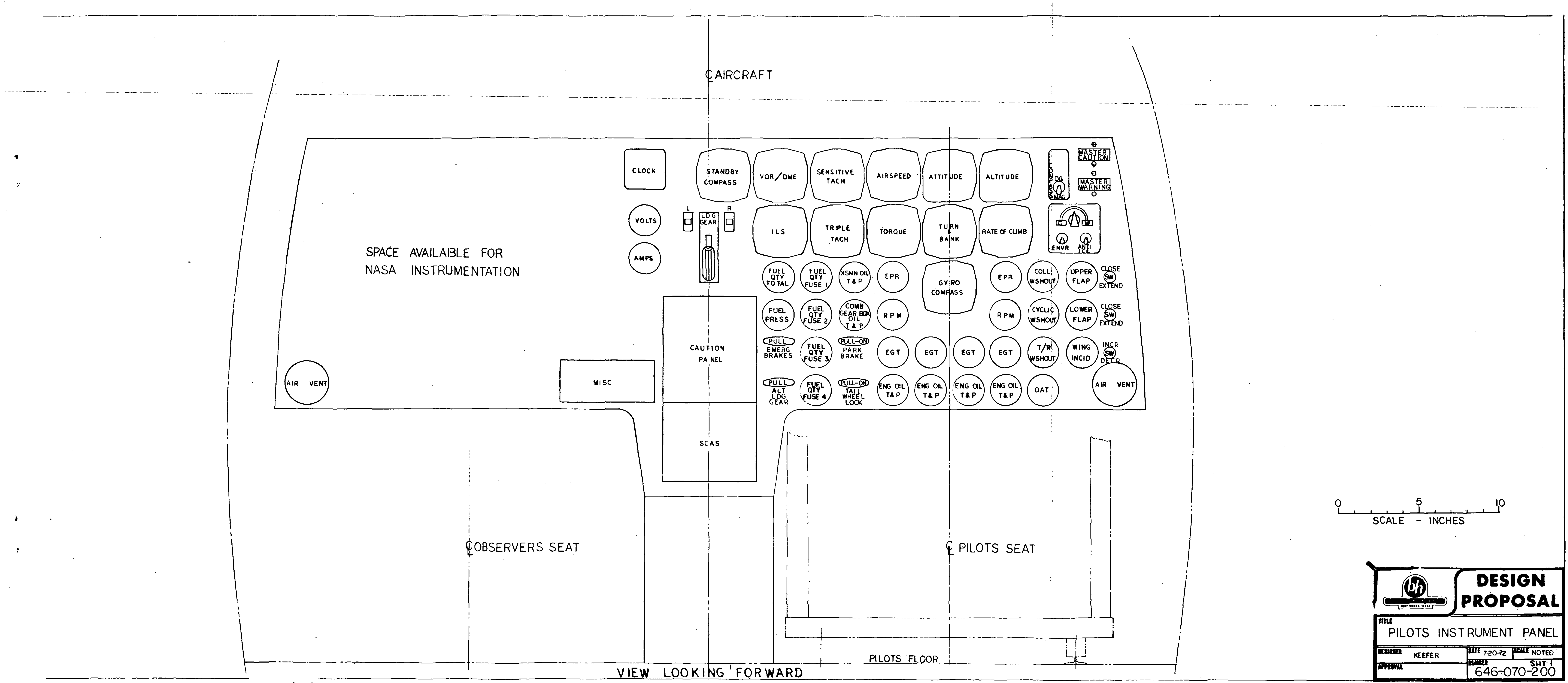

SECTION B-B
SECTION A-A
CASTOR LOCK
TAIL WHEEL CASTOR LOCKING CABLE
PNEUMATIC CENTERING CYLINDER
STRUT HOUSING
INNER AIR-OIL SHOCK ABSORBING STRUT
ORIFICE
OIL RESERVOIR
PISTON
AIR RESERVOIR
STATIONARY DORSAL FIN
FIBERGLASS FAIRING (PIVOTS WITH OUTER STRUT)
OUTER STRUT
TRAILING ARM
16 x 4.4 x 10 PLY TUBELESS TIRE STATIC POSITION
12" AXLE STROKE
0 1 2 3 4 5 6 7 8 9 10
SCALE (INCHES)
SIDE VIEW
SCALE 1/2
DESIGN PROPOSAL

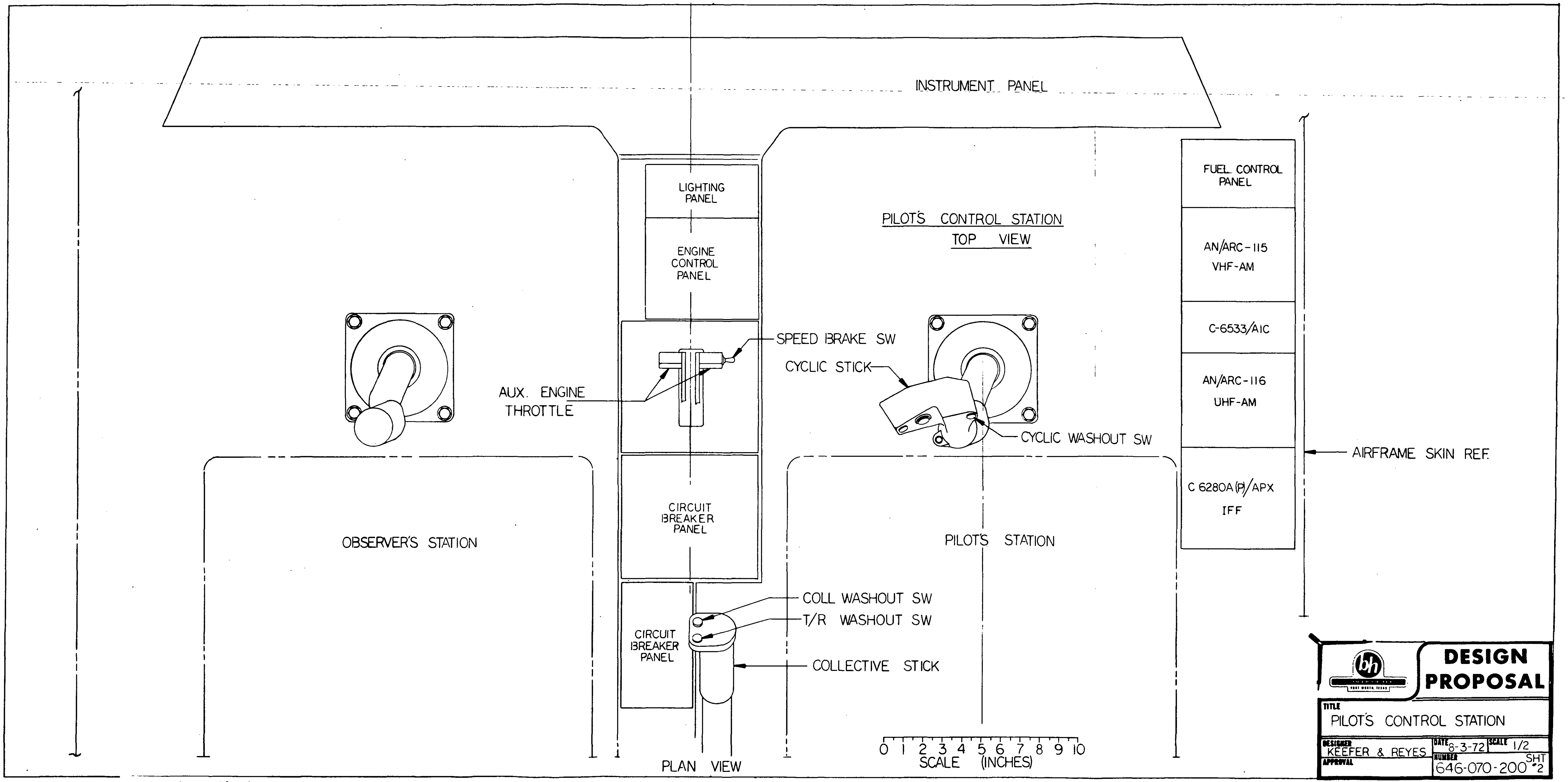


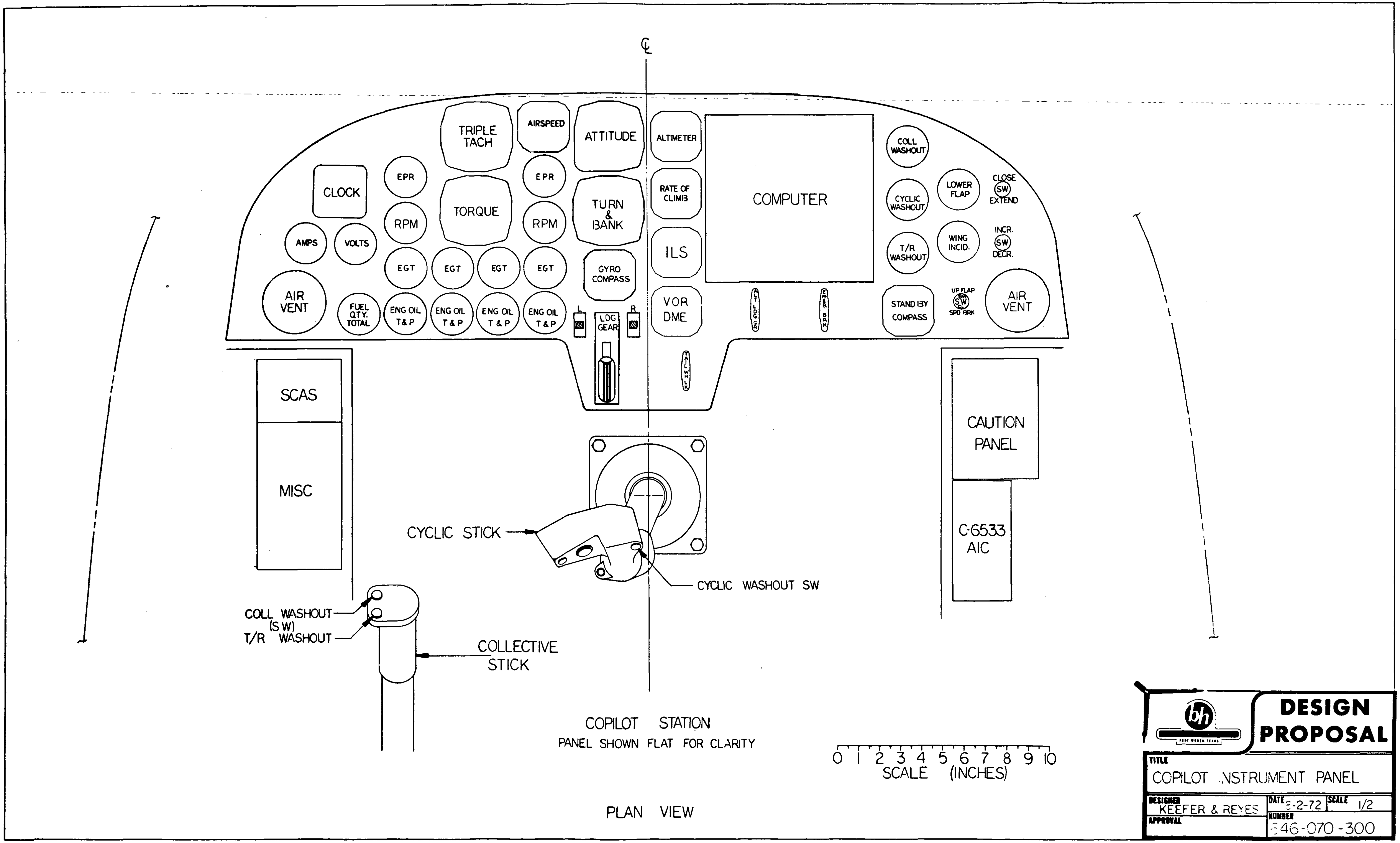
DESIGN PROPOSAL

TITLE INSTALLATION AUXILIARY ENGINE MOUNT

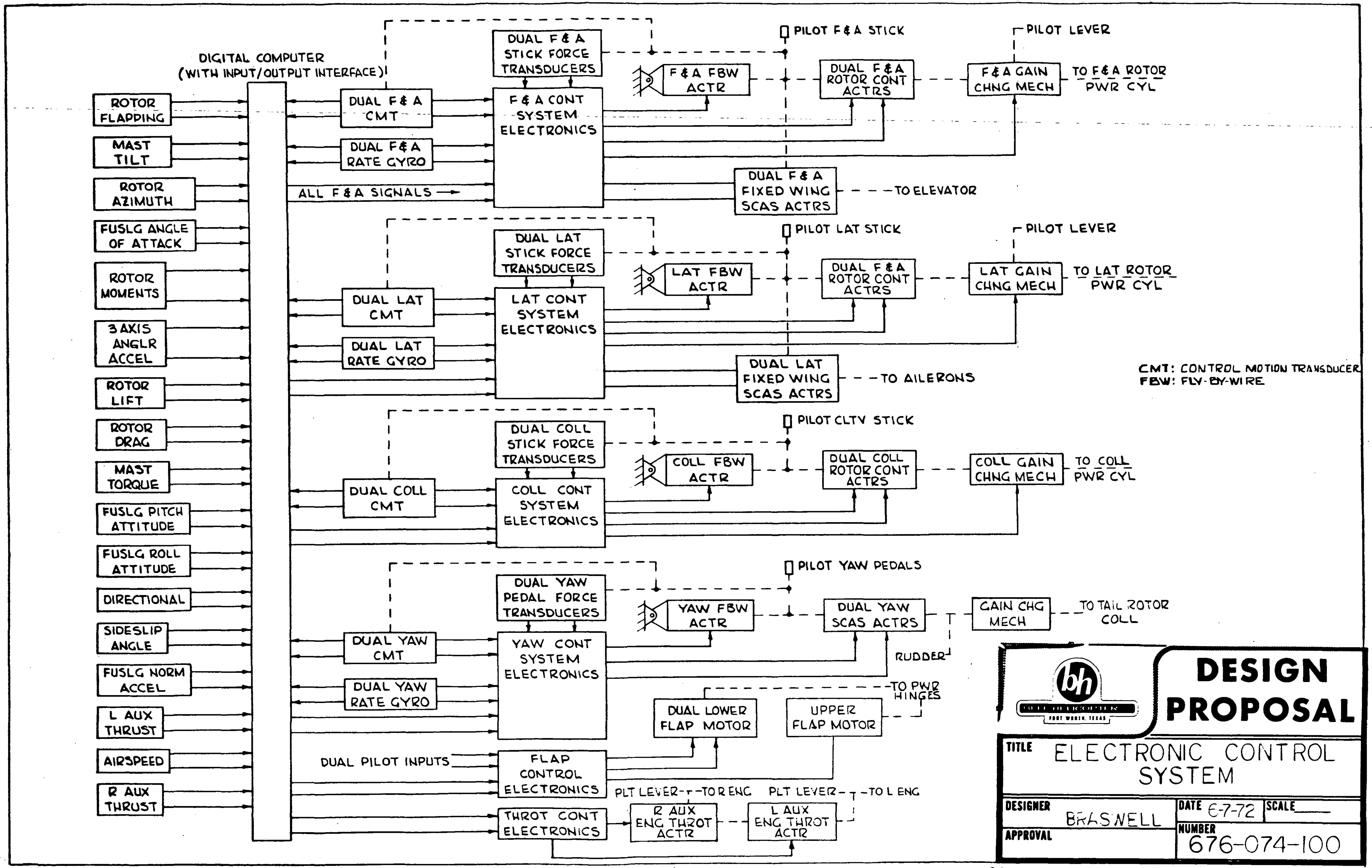
DESIGNED | DATE | SCALE
APPROVAL | NUMBER 646-060-200

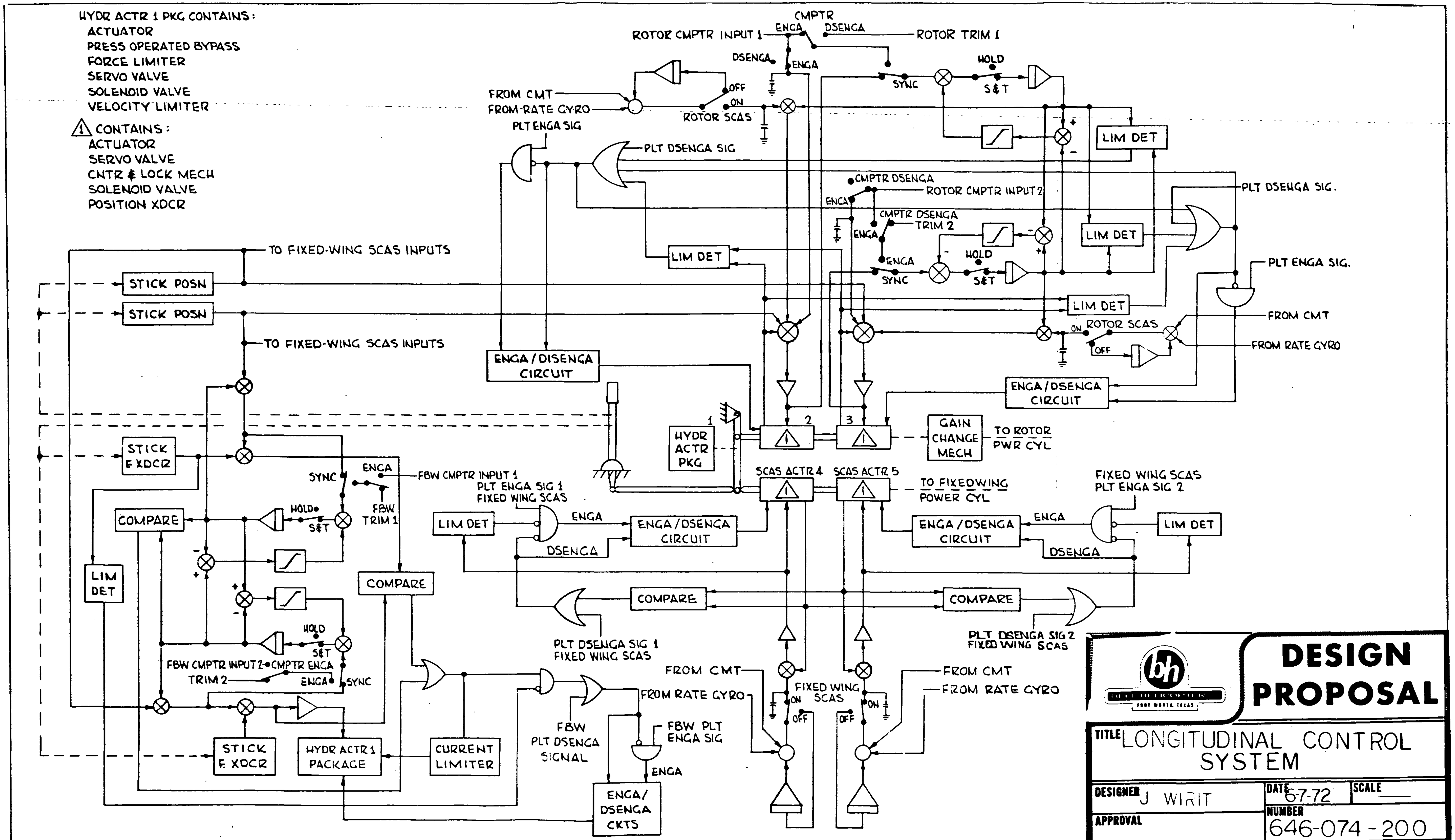

℄ AIRCRAFT
SPACE AVAILABLE FOR NASA INSTRUMENTATION
CLOCK
STANDBY COMPASS
VOR/DME
SENSITIVE TACH
AIRSPEED
ATTITUDE
ALTITUDE
MASTER CAUTION
MASTER WARNING
VOLTS
AMPS
L
R
LDG GEAR
ILS
TRIPLE TACH
TORQUE
TURN & BANK
RATE OF CLIMB
ENVR
ANTI ICE
FUEL QTY TOTAL
FUEL QTY FUSE 1
XSMN OIL T&P
EPR
GYRO COMPASS
COLL WSHOUT
UPPER FLAP
CLOSE SW EXTEND
FUEL PRESS
FUEL QTY FUSE 2
COMB GEAR BOX OIL T&P
RPM
CYCLIC WSHOUT
LOWER FLAP
CAUTION PANEL
PULL EMERG BRAKES
FUEL QTY FUSE 3
PULL-ON PARK BRAKE
EGT
T/R WSHOUT
WING INCID
INCR SW DECR
AIR VENT
MISC
PULL ALT LDG GEAR
FUEL QTY FUSE 4
PULL-ON TAIL WHEEL LOCK
ENG OIL T&P
OAT
SCAS
℄ OBSERVERS SEAT
℄ PILOTS SEAT
PILOTS FLOOR
VIEW LOOKING FORWARD
0 5 10
SCALE - INCHES
DESIGN PROPOSAL
TITLE
PILOTS INSTRUMENT PANEL
DESIGNER KEEFER
DATE 7-20-72
SCALE NOTED
APPROVAL
NUMBER 646-070-200
SHT 1


INSTRUMENT PANEL
LIGHTING PANEL
ENGINE CONTROL PANEL
SPEED BRAKE SW
CYCLIC STICK
AUX. ENGINE THROTTLE
CYCLIC WASHOUT SW
PILOT'S CONTROL STATION
TOP VIEW
FUEL CONTROL PANEL
AN/ARC-115 VHF-AM
C-6533/AIC
AN/ARC-116 UHF-AM
C 6280A(P)/APX IFF
AIRFRAME SKIN REF.
CIRCUIT BREAKER PANEL
OBSERVER'S STATION
PILOT'S STATION
COLL WASHOUT SW
T/R WASHOUT SW
CIRCUIT BREAKER PANEL
COLLECTIVE STICK
0 1 2 3 4 5 6 7 8 9 10
SCALE (INCHES)
PLAN VIEW
DESIGN PROPOSAL
TITLE
PILOT'S CONTROL STATION
DESIGNER
KEEFER & REYES
APPROVAL
DATE 8-3-72
SCALE 1/2
NUMBER
646-070-200
SHT 2



TRIPLE TACH
AIRSPEED
ATTITUDE
ALTIMETER
COMPUTER
COLL WASHOUT
CLOCK
EPR
EPR
TORQUE
RPM
RPM
TURN & BANK
RATE OF CLIMB
CYCLIC WASHOUT
LOWER FLAP
CLOSE SW EXTEND
AMPS
VOLTS
EGT
EGT
EGT
EGT
GYRO COMPASS
ILS
T/R WASHOUT
WING INCID.
INCR. SW DECR.
AIR VENT
FUEL QTY. TOTAL
ENG OIL T&P
ENG OIL T&P
ENG OIL T&P
ENG OIL T&P
L
LDG GEAR
R
VOR DME
STAND BY COMPASS
UP FLAP SW SPD BRK
AIR VENT
SCAS
MISC
CAUTION PANEL
C-6533 AIC
CYCLIC STICK
CYCLIC WASHOUT SW
COLL WASHOUT
(S W)
T/R WASHOUT
COLLECTIVE STICK
COPILOT STATION
PANEL SHOWN FLAT FOR CLARITY
0 1 2 3 4 5 6 7 8 9 10
SCALE (INCHES)
PLAN VIEW
DESIGN PROPOSAL
TITLE
COPILOT INSTRUMENT PANEL
DESIGNER
KEEFER & REYES
DATE 8-2-72
SCALE 1/2
APPROVAL
NUMBER
646-070-300


DIGITAL COMPUTER (WITH INPUT/OUTPUT INTERFACE)
ROTOR FLAPPING
MAST TILT
ROTOR AZIMUTH
FUSLG ANGLE OF ATTACK
ROTOR MOMENTS
3 AXIS ANGLR ACCEL
ROTOR LIFT
ROTOR DRAG
MAST TORQUE
FUSLG PITCH ATTITUDE
FUSLG ROLL ATTITUDE
DIRECTIONAL
SIDESLIP ANGLE
FUSLG NORM ACCEL
L AUX THRUST
AIRSPEED
R AUX THRUST
DUAL F & A CMT
DUAL F & A RATE GYRO
ALL F & A SIGNALS
DUAL F & A STICK FORCE TRANSDUCERS
F & A CONT SYSTEM ELECTRONICS
F & A FBW ACTR
PILOT F & A STICK
DUAL F & A ROTOR CONT ACTRS
PILOT LEVER
F & A GAIN CHNG MECH
TO F & A ROTOR PWR CYL
DUAL F & A FIXED WING SCAS ACTRS
TO ELEVATOR
DUAL LAT CMT
DUAL LAT RATE GYRO
DUAL LAT STICK FORCE TRANSDUCERS
LAT CONT SYSTEM ELECTRONICS
LAT FBW ACTR
PILOT LAT STICK
DUAL F & A ROTOR CONT ACTRS
PILOT LEVER
LAT GAIN CHNG MECH
TO LAT ROTOR PWR CYL
DUAL LAT FIXED WING SCAS ACTRS
TO AILERONS
CMT: CONTROL MOTION TRANSDUCER
FBW: FLY-BY-WIRE
DUAL COLL CMT
DUAL COLL STICK FORCE TRANSDUCERS
COLL CONT SYSTEM ELECTRONICS
COLL FBW ACTR
PILOT CLTV STICK
DUAL COLL ROTOR CONT ACTRS
COLL GAIN CHNG MECH
TO COLL PWR CYL
DUAL YAW CMT
DUAL YAW RATE GYRO
DUAL YAW PEDAL FORCE TRANSDUCERS
YAW CONT SYSTEM ELECTRONICS
YAW FBW ACTR
PILOT YAW PEDALS
DUAL YAW SCAS ACTRS
GAIN CHG MECH
TO TAIL ROTOR COLL
RUDDER
TO PWR HINGES
DUAL LOWER FLAP MOTOR
UPPER FLAP MOTOR
DUAL PILOT INPUTS
FLAP CONTROL ELECTRONICS
THROT CONT ELECTRONICS
PLT LEVER
TO R ENG
R AUX ENG THROT ACTR
PLT LEVER
TO L ENG
L AUX ENG THROT ACTR
DESIGN PROPOSAL
TITLE ELECTRONIC CONTROL SYSTEM
DESIGNER BRASWELL
DATE 6-7-72
SCALE
APPROVAL
NUMBER 676-074-100


HYDR ACTR 1 PKG CONTAINS:
ACTUATOR
PRESS OPERATED BYPASS
FORCE LIMITER
SERVO VALVE
SOLENOID VALVE
VELOCITY LIMITER
CONTAINS:
ACTUATOR
SERVO VALVE
CNTR & LOCK MECH
SOLENOID VALVE
POSITION XDCR
CMPTR
ROTOR CMPTR INPUT 1
ENGA
DSENGA
ROTOR TRIM 1
HOLD
SYNC
S&T
FROM CMT
FROM RATE GYRO
OFF
ON
ROTOR SCAS
PLT ENGA SIG
PLT DSENGA SIG
LIM DET
CMPTR DSENGA
ROTOR CMPTR INPUT 2
TRIM 2
PLT DSENGA SIG.
PLT ENGA SIG.
TO FIXED-WING SCAS INPUTS
STICK POSN
STICK POSN
TO FIXED-WING SCAS INPUTS
ENGA / DISENGA CIRCUIT
ENGA / DSENGA CIRCUIT
HYDR ACTR PKG
GAIN CHANGE MECH
TO ROTOR PWR CYL
STICK F XDCR
FBW CMPTR INPUT 1
FBW TRIM 1
PLT ENGA SIG 1 FIXED WING SCAS
SCAS ACTR 4
SCAS ACTR 5
TO FIXEDWING POWER CYL
FIXED WING SCAS PLT ENGA SIG 2
COMPARE
LIM DET
ENGA / DSENGA CIRCUIT
DSENGA
COMPARE
PLT DSENGA SIG 1 FIXED WING SCAS
PLT DSENGA SIG 2 FIXED WING SCAS
FBW CMPTR INPUT 2
CMPTR ENGA
FROM CMT
FIXED WING SCAS
FROM RATE GYRO
FBW PLT DSENGA SIGNAL
FBW PLT ENGA SIG
STICK F. XDCR
HYDR ACTR 1 PACKAGE
CURRENT LIMITER
ENGA / DSENGA CKTS
DESIGN PROPOSAL
TITLE LONGITUDINAL CONTROL SYSTEM
DESIGNER J WIRIT
DATE 6-7-72
SCALE
APPROVAL
NUMBER 646-074-200